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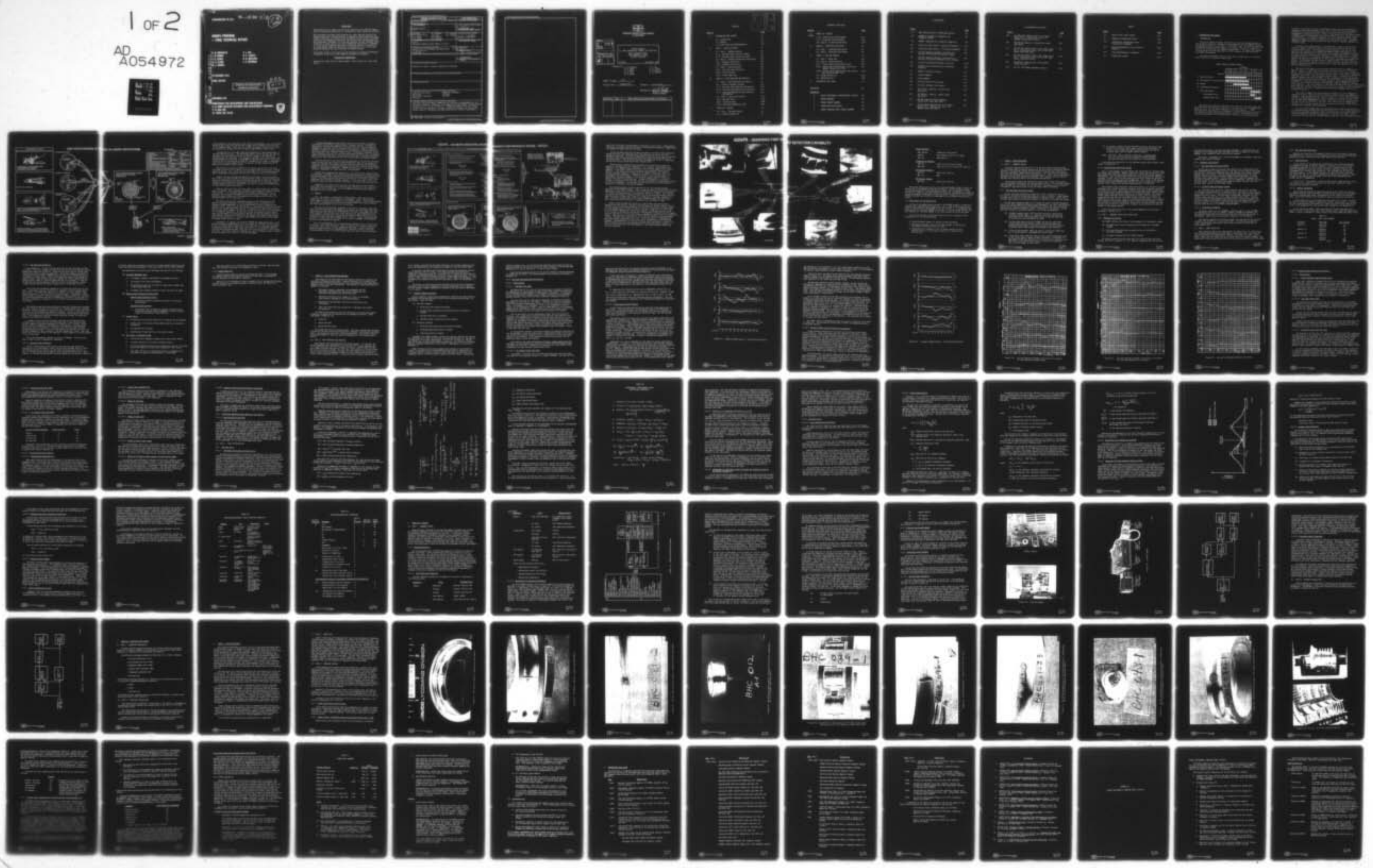

USAAVRADCOM TR 78-5

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# AIDAPS PROGRAM
# – FINAL TECHNICAL REPORT


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24 OCTOBER 1974

FINAL REPORT

APPROVED FOR PUBLIC RELEASE
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PREPARED FOR


DIRECTORATE FOR DEVELOPMENT AND ENGINEERING
U. S. ARMY AVIATION RESEARCH AND DEVELOPMENT COMMAND
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This Final Technical Report summarizes the technical accomplishments on the Automatic Inspection, Diagnostic and Prognostic System (AIDAPS) Program during the period of 27 Jun 73 to 30 Nov 76. AIDAPS was intended for use in Army Aircraft and was intended to be used to reduce maintenance cost and improve flight safety by continuous in-flight monitoring of aircraft subsystems.

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FINAL TECHNICAL REPORT
CDRL SEQUENCE NO. A018
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Program Manager

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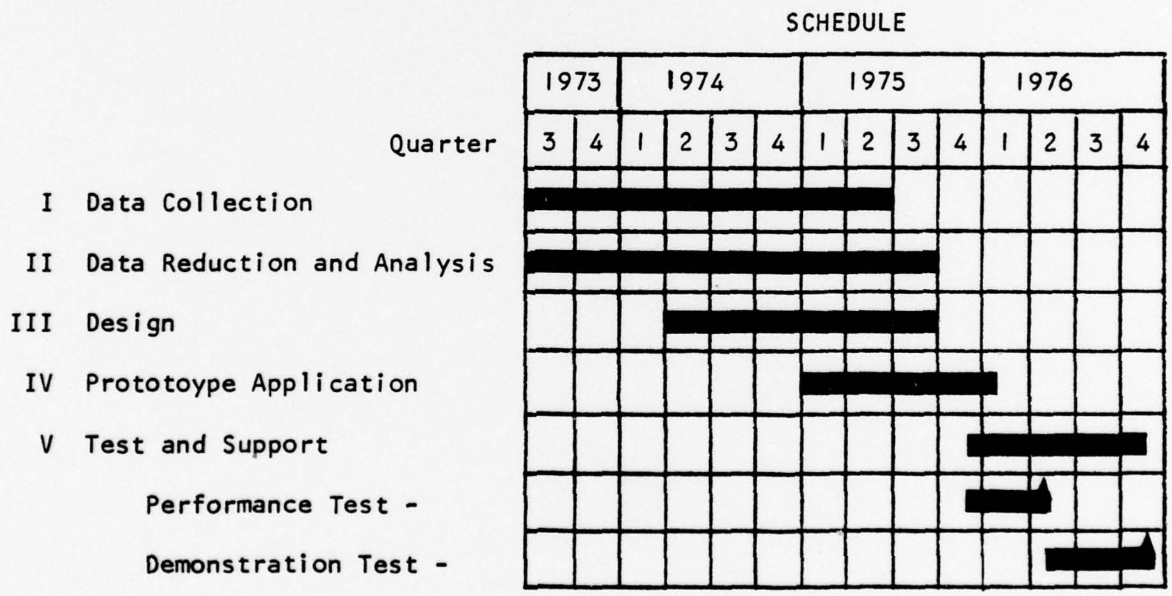
# 1. INTRODUCTION AND SUMMARY

## 1.1 INTRODUCTION

This Final Technical Report summarizes the technical accomplishments on the Automatic Inspection, Diagnostic, and Prognostic System (AIDAPS) program during the period of 27 June 1973 to 30 November 1976. The work was performed by AiResearch Manufacturing Company, a Division of The Garrett Corporation, under U. S. Army Contract No. DAAJ01-73-C-0374(P6C). The report was prepared in conformance with the requirements of CDRL Sequence No. A018 and DID No. DI-S-1800, and Addendum.

The system development and test tasks were accomplished in five phases over a period of 41 months as shown below:

AIDAPS PROGRAM ACCOMPLISHMENTS




The collection, reduction and analysis of test cell and flight test data for a period of over two years resulted in an effective universal AIDAPS fault detection logic for the major components on Army helicopters. These fault detection techniques were implemented in prototype hardware and tested in over 200 test flights during approximately 11 months. Performance testing served the purpose of refining preliminary limits, for the UH-1H test bed

aircraft in particular, as well as prototype system check out and debugging. Subsequent Army-conducted demonstration tests further verified the successful achievement of the prime technical objective of detecting the presence of degraded parts automatically and in flight. Note that most of the parts detected by AIDAPS could not be detected during flight operations using normal aircraft instrumentation. Furthermore, subsequent cell tests demonstrated that these parts were capable of additional operation without total failure. Therefore, AIDAPS can detect faulty parts before the part condition becomes hazardous to aircraft flight.

In order to put these benefits in perspective in terms of quantitative financial rewards, AiResearch sponsored a cost effectiveness study for the UH-IH and AAH aircraft. The UH-IH was selected because it was the aircraft used as a test bed during the AIDAPS development program and because a large data base of maintenance cost information was available from TAMMS which could be used to evaluate the cost effectiveness of an AIDAPS application. The AAH was selected since it still offers the opportunity to achieve the full benefits possible by applying the AIDAPS during the aircraft development and production phases. In both cases the application of AIDAPS proved to be cost effective. However, the financial rewards in terms of the return-on-investment and a short period to achieve a positive-cash-flow condition make the application to the AAH far more attractive.

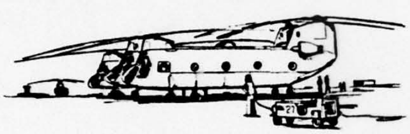
The cost effectiveness study indicated that application of AIDAPS to the AAH will result in significant cost savings, $218 million net in a ten-year period. Figure I-I depicts the results of this study. Four areas of potential savings were included in the cost effectiveness model--flight safety, operational readiness, parts costs, and maintenance man-hour costs. A description of the computer program which was developed is given in Appendix A. Army maintenance data for the UH-IH were used to formulate the basic model; see references cited in Appendix A. Pertinent data for those components monitored were reviewed in order that only those costs which would have been saved by AIDAPS be included in the analysis. Thus, for example, in the case of accidents and aborts, those events which were due to causes other than material failure, as well as those which were due to material failures in components not monitored by AIDAPS, were not considered for savings.

The study of AIDAPS cost effectiveness was based on certain assumptions regarding aircraft fleet size and utilization, equipment costs and AIDAPS performance. These assumptions are listed in Figure I-I. Using these assumptions and the Army maintenance data for the UH-IH results in the cost savings shown for the four major AIDAPS benefit areas. These savings add to a total savings of $132 million for the UH-IH. The total life cycle costs for AIDAPS application to this aircraft are $76 million, leaving a net savings of $56 million accrued over a ten-year operating period.

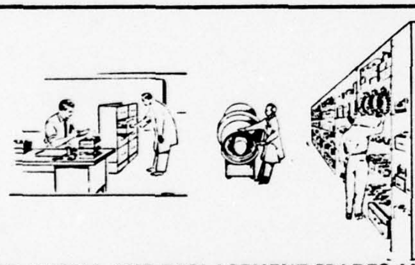
The cost effectiveness model was also applied to the AAH by extrapolation of maintenance data and applying the assumptions shown. The different proportion of cost savings among the benefit areas is affected by the different number of monitored components (engines and gearboxes) on the AAH, the higher initial cost of the aircraft and the improved safety reliability due to redundant engines.

## AIDAPS BENEFIT AREAS

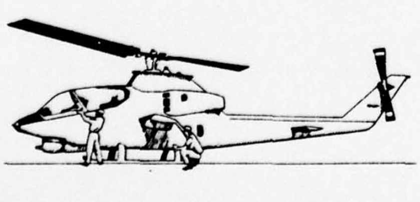
- FLIGHT SAFETY . . . BY REDUCING ACCIDENTS DUE TO COMPONENT FAILURE AND UNNECESSARY FLIGHT ABORTS



- OPERATIONAL READINESS

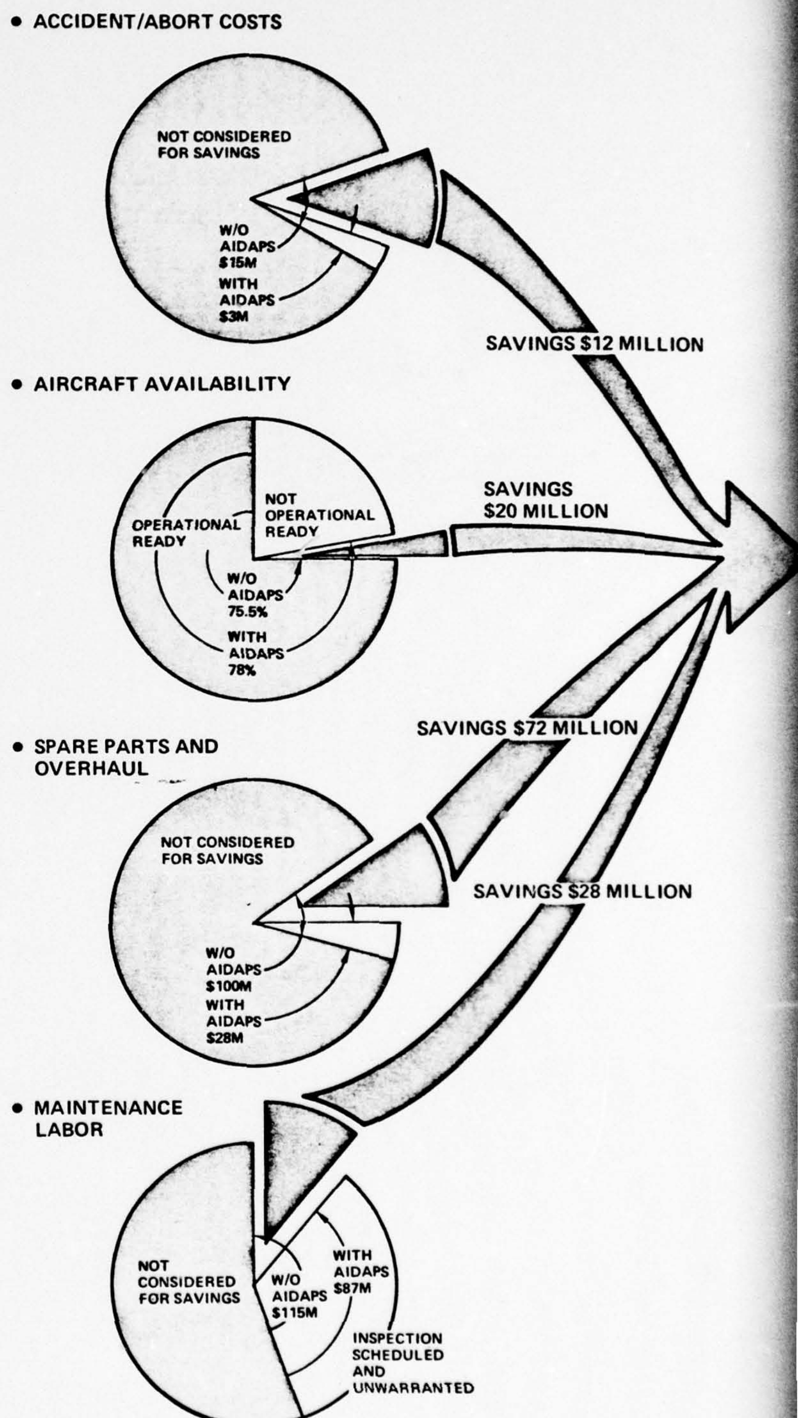


- REDUCED INITIAL AND REPLACEMENT SPARES AND OVERHAUL . . . BY REDUCING SCHEDULED AND UNWARRANTED REMOVALS, AND SECONDARY DAMAGE



- REDUCED MAINTENANCE . . . BY REDUCING TROUBLE SHOOTING AND MAINTENANCE TIME AND IMPROVING SCHEDULING OF AIRCRAFT MAINTENANCE AND REPAIR

# COST EFFECTIVENESS OF



# ESS OF AIDAPS APPLICATIONS

ASSUMPTIONS

| | UH-1H | AAH |
|---|---|---|
| • FLEET SIZE | 2000 A/C | 500 A/C |
| • AIRCRAFT COST | $400,000 | $2.24 MILLION |
| • MMH/FH | 3.9 | 7.8 |
| • INITIAL AIRCRAFT AVAILABILITY | 0.755 | 0.8 |
| • USAGE | 20 HRS/MONTH | 20 HRS/MONTH |
| • AIDAPS DETECTION ACCURACY | 0.85 | 0.85 |
| • FALSE ALARM RATE | 0.5/1000 FLT HRS | 0.5/1000 FLT HRS |
| • AIRCRAFT OPERATION PERIOD | 10 YEARS | 10 YEARS |
| • MAINTENANCE MAN HOUR COST | $13.28/HR | $13.28/HR |
| • ACHIEVES ON-CONDITION MAINTENANCE OF AIRCRAFT | | |

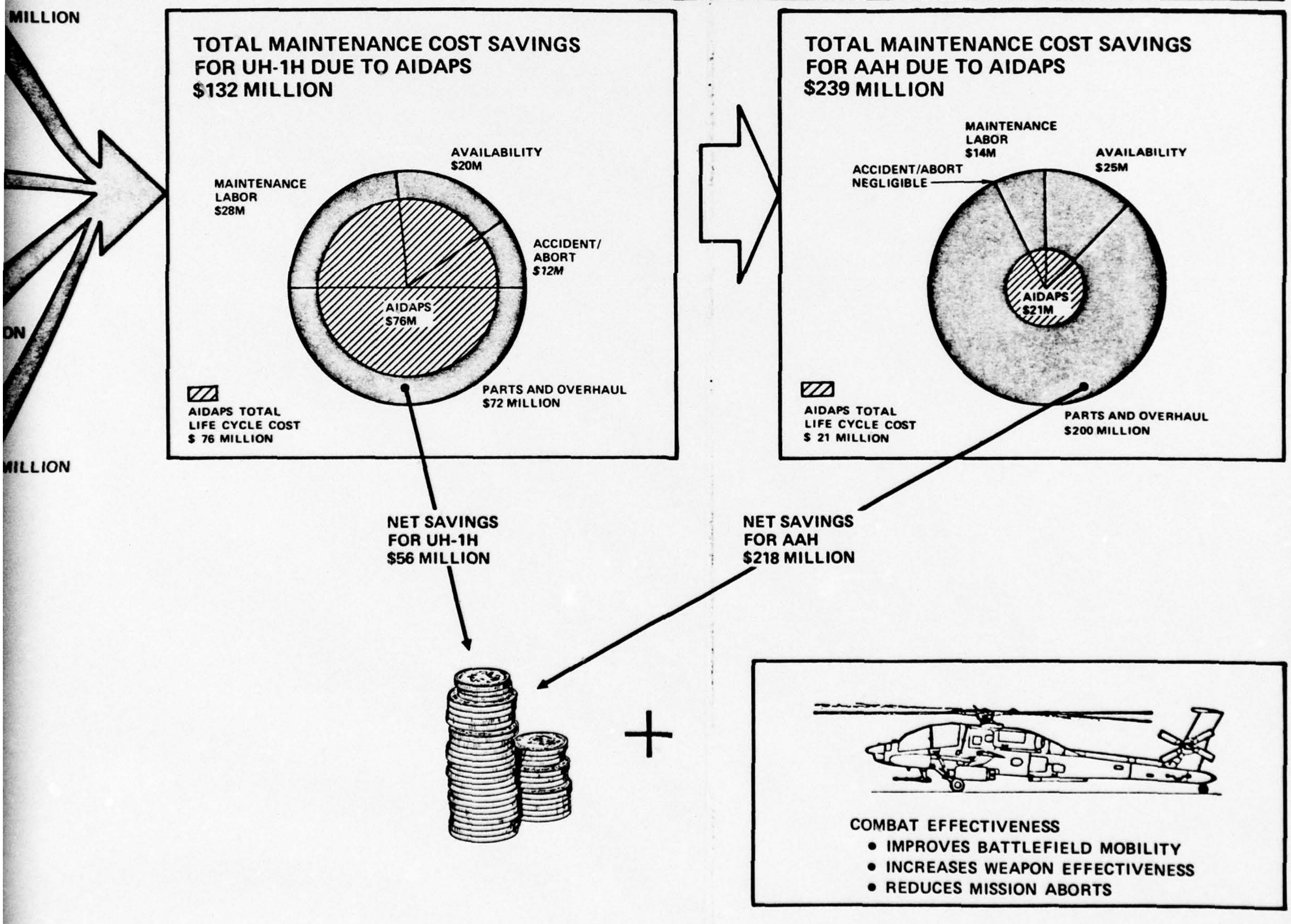



FIGURE 1-1



2

The extrapolation was conservative and several key parameters, e.g., maintenance man-hours per flight hour and aircraft usage, were estimated on the low side of the expected value. Note that an increase in usage of 10 hours per aircraft per month would increase the cost savings by approximately $100 million.

The application to the AAH involved extrapolation of maintenance data of the mature UH-1H aircraft. The additional benefit which would be derived from improved fault detection during the reliability growth of the AAH and its subsystems was not estimated. In the case of the AAH or any new aircraft, the maximum benefits are derived if the AIDAPS is deployed from initial production. A portion of this savings was assumed for purposes of this model in that application during initial production was assumed for the AAH, whereas the UH-1H application included retrofit costs.

The total cost savings of $239 million results from a total life cycle investment of $21 million, leaving a net savings of $218 million over a 10-year operating period. The initial cost for procurement of aircraft-mounted equipment is estimated at 65 percent of the total life-cycle cost for the fleet.

Some of the cost savings associated with the benefit areas of accident/abort costs, aircraft availability and maintenance labor are controversial and may not be considered to be totally available as savings by the Army. Notwithstanding, the estimated savings expected for the AAH in the spare parts and overhaul benefits alone are significant, and offer a return of investment early in the aircraft operational period.

Clearly, there are other than cost benefits to be derived from the application of AIDAPS. These include crew safety, maintenance efficiency and combat effectiveness. The increased combat effectiveness, although difficult to evaluate, can increase the probability of putting Army fire power into the battlefield when needed, and then complete the assigned attack mission with reduced probability of mission abort.

Considering the current government posture of maximizing results and benefits while simultaneously minimizing expenditures, the adoption of a program of AIDAPS deployment on initial AAH production could indeed be considered timely. Similarly, with the advent of a new generation of Army aircraft, specifically the AAH and UTTAS, and the life extension of such proven craft as the UH-1 and CH-47, emphasis on quality, reliability and facility of maintenance is of paramount importance. As the cost of each new production craft grows, more concrete means of providing dollar savings and tangible benefits must be investigated. Exploring the results of the current cost effectiveness study, even on a cursory basis, it is evident that AIDAPS can provide a viable means of accomplishing just such a goal.

Throughout its rigorous test program, AIDAPS has demonstrated that it can effectively detect degradations due to the excessive vibrations of moving parts as well as detecting performance efficiency reductions in the engine gas path. It should be noted that in many cases the severity of the degraded component installed and detected was many times less than that called for in the original test plan. This merely shows the adaptability that the AIDAPS system is capable of and the sensitivity to which it can be programmed.

Several advantages can obviously be seen with an automatic diagnostic system detecting imminent trouble before it actually occurs. On-condition maintenance of monitored components can be achieved and resultant benefits subsequently realized, such as increased flight safety, longer time between overhauls, greater aircraft availability, and especially, reduced supplies of spare parts and their related inventory costs. Furthermore, automatic tracking of component condition will provide maintenance personnel with a prognostic capability allowing an easier and more productive means of scheduling maintenance actions that cannot be attained with a conventional maintenance method.

With the advent of new flight profile techniques in the military aviation arena (witness the new reliance on nap-of-the-earth, NOE, flying versus the traditional method), more severe and different wear patterns are manifesting themselves and parts are deteriorating at a faster rate, thereby necessitating changes in maintenance procedures. AIDAPS has proven that it can detect failing parts independent of flight profiles, including NOE.

Several Army commands have stated that, if a system could indeed perform as described, increased flight safety and reduced maintenance could be achieved and improved operational readiness and combat effectiveness would necessarily follow. Of primary concern was the high incidence of "no-trouble-found" when components were returned from depot overhaul. It was felt that if only 10 percent of the occurrences could be eliminated, the system could be justified.

Coupled with the ever increasing cost and complexity of new aircraft, it would seem that the adoption of a service-wide requirement for on-condition maintenance would be opportune, and that AIDAPS could be the best vehicle available to fulfill this goal.

## 1.2 SUMMARY

This final report is organized by program phase. Each major section summarizes the results of one phase of the program. Figure 1-2 provides a brief summary of the accomplishments of the AIDAPS program. These accomplishments are placed in perspective by showing the overall objectives of such a system and the projected benefits which will be realized.

The data collection phase, discussed in Section 2, provided a large data base from which to develop the AIDAPS system concept and fault detection logic. These data included reports and recorded data from other development test and study programs, Army manuals, and flight and test cell data generated during this program. The test cell testing was performed by AVCO Lycoming Division (engines) and Bell Helicopter Company (drive trains). Flight testing was done by the U. S. Army Aviation Test Board at Ft. Rucker, Alabama, with Hawthorne Aviation as maintenance contractor.

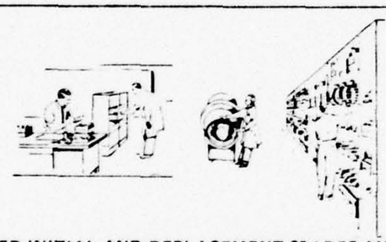
Four UH-1H helicopters were instrumented by AiResearch for in-flight data collection. Over 800 flights were flown, 70 percent with implants installed. In the test cells, and in flight, components were used in newly overhauled condition (baseline) and with degraded parts (implants) actually installed. Two-hundred-forty reels of 14-track analog tape were used to record vibration

# AIDAPS . . . AUTOMATIC INSPECTION, DIAGNO

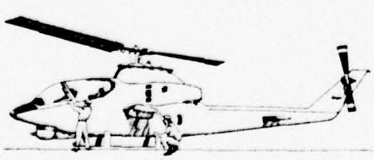
## AIDAPS OBJECTIVES

• FLIGHT SAFETY . . . BY REDUCING ACCIDENTS DUE TO COMPONENT FAILURE AND UNNECESSARY FLIGHT ABORTS

• INCREASED OPERATIONAL READINESS



• REDUCED INITIAL AND REPLACEMENT SPARES AND OVERHAUL . . . BY REDUCING SCHEDULED AND UNWARRANTED REMOVALS, AND SECONDARY DAMAGE



• REDUCED MAINTENANCE . . . BY REDUCING TROUBLE SHOOTING AND MAINTENANCE TIME AND IMPROVING SCHEDULING OF AIRCRAFT MAINTENANCE AND REPAIR

GARRETT

AIRESEARCH
MANUFACTURING CO
OF CALIFORNIA

A DIVISION OF THE GARRETT CORPORATION

## PROGRAM ACCOMPLISHMENTS

PHASE I – DATA COLLECTION

- DESIGNED DATA COLLECTION SYSTEM FOR UH-1H
- INSTRUMENTED FOUR UH-1H HELICOPTERS
- FLEW 825 DATA COLLECTION FLIGHTS
- GENERATED DATA BASE FOR FAULT DETECTION LOGIC DEVELOPMENT

PHASE II – DATA REDUCTION AND ANALYSIS

- CONVERTED THE DATA BASE INTO A USEFUL INFORMATIONAL FORM AND PERFORMED THE ANALYTICAL DEVELOPMENT OF THE FAULT DETECTION LOGIC

PHASE III – DESIGN

- SYSTEM DESIGNED FOR UNIVERSAL APPLICATION TO ARMY HELICOPTERS
- PROTOTYPE HARDWARE DESIGNED TO IMPLEMENT FAULT DETECTION LOGIC ON UH-1H TEST VEHICLE
- HYBRID I SYSTEM CONFIGURATION OFFERS IMMEDIATE ON-AIRCRAFT FAULT COMMUNICATION
- HYBRID II SYSTEM CONFIGURATION OFFERS CONTINUOUS AIRBORNE DATA RECORDING FOR POST FLIGHT ANNUNCIATION AT GROUND SEGMENT

PHASE IV – PRO

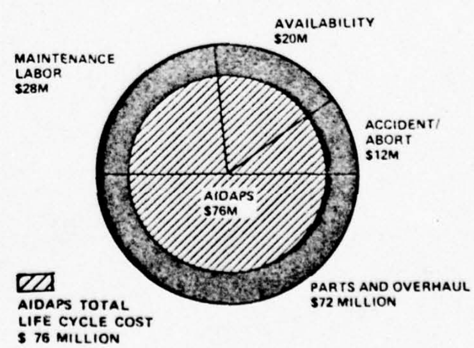
- EQUIPPED SEGMENTS CONFIGUR
  - TWO H
  - TWO H
- PROVIDED
- PROVIDED

PHASE V – TES

- BENCH TES
  - STATI
  - ENVIR
- AIRBORNE
  - FLIGH
  - PERFO
  - DEMO
- DEMONSTI
  - 163 IM
  - 120 IM
  - 2.6% F
- PROJECTE
  - AIRBO
  - GROU
  - FALSE
  - FAUL

## AIDAPS BENEFITS . . .

TOTAL MAINTENANCE COST SAVINGS FOR UH-1H DUE TO AIDAPS
$132 MILLION




TOTAL
FOR A
$239 M

ACCID
NEGLI

AIDAPS
LIFE CY
$ 21 MIL

# IAGNOSTIC AND PROGNOSTIC SYSTEM... AIDAPS

**ISHMENTS**

**HASE IV – PROTOTYPE APPLICATION**

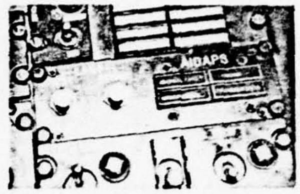
- EQUIPPED FOUR UH-1H WITH AIRBORNE SEGMENTS OPERABLE IN EITHER HYBRID I OR II CONFIGURATION
  - – TWO HYBRID I CONFIGURATIONS
  - – TWO HYBRID II CONFIGURATIONS
- PROVIDED TWO GROUND SEGMENTS
- PROVIDED GROUND SUPPORT EQUIPMENT

**HASE V – TEST AND SUPPORT**

- BENCH TESTS
  - – STATIC SYSTEM TESTS
  - – ENVIRONMENTAL TESTS
- AIRBORNE TESTS
  - – FLIGHT SAFETY QUALIFICATION
  - – PERFORMANCE IMPLANT TESTS
  - – DEMONSTRATION IMPLANT TESTS
- DEMONSTRATION TEST RESULTS
  - – 163 IMPLANT TESTS SCORED
  - – 120 IMPLANTS SUCCESSFULLY DETECTED
  - – 2.6% FALSE ALARM RATE EXPERIENCED
- PROJECTED SYSTEM PERFORMANCE
  - – AIRBORNE SEGMENT MTBF – 1000 HOURS
  - – GROUND SEGMENT MTBF – 880 HOURS
  - – FALSE ALARM RATE – 1/2000 HOURS
  - – FAULT DETECTION RATE – 85%

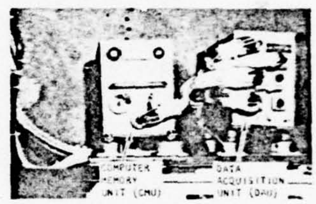
RESULTS

**AIDAPS...** HAS THE ABILITY TO DETECT THE PRESENCE OF DEGRADED PARTS IN FAULTY COMPONENTS DURING NORMAL AIRCRAFT FLIGHT OPERATIONS . . .

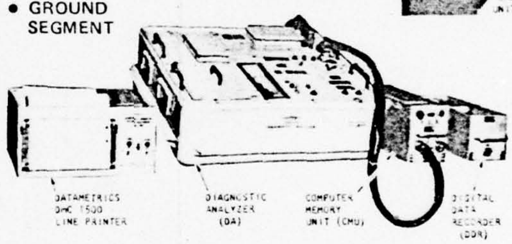



- FLIGHT STATUS PANEL/COCKPIT DISPLAY

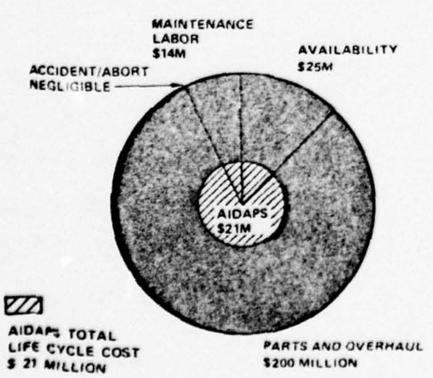
- AIRBORNE SEGMENT




- GROUND SEGMENT

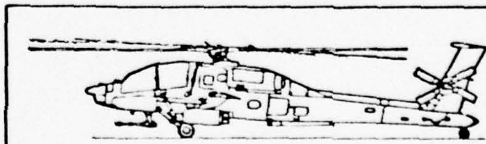



**TOTAL MAINTENANCE COST SAVINGS FOR AAH DUE TO AIDAPS $239 MILLION**




NET SAVINGS FOR UH-1H $56 MILLION

NET SAVINGS FOR AAH $218 MILLION



- COMBAT EFFECTIVENESS . . . IMPROVE BATTLEFIELD MOBILITY BY INCREASING AIRCRAFT AVAILABILITY AND REDUCING MISSION ABORTS

FIGURE 1-2



2

data from 15 different accelerometers installed on the aircraft. Approximately 500 hours of digital data were recorded including engine pressures, temperatures and speeds and supporting aircraft data.

During the data reduction and analysis phase, a universal fault detection concept and logic was developed which functions well during all aircraft operating conditions. This was accomplished by analysis of the data to develop the basic logic and determine which conditions affect the detection and/or false alarm rate. The system was designed to compensate for (or in some cases exclude data during) these conditions, automatically.

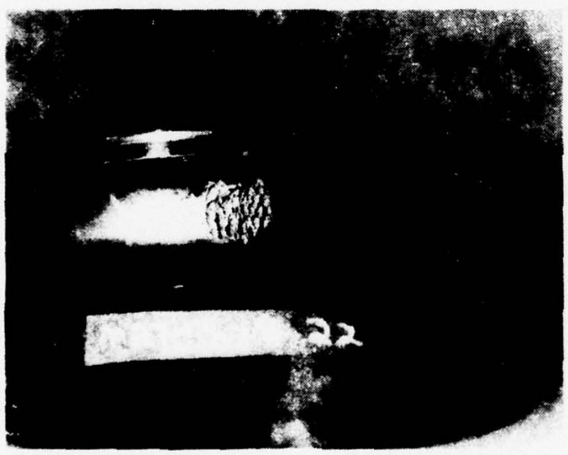
The prototype hardware design was based on providing fault detection capability for major aircraft components, i.e., engines, transmissions and gearboxes. The AIDAPS hardware consists of an airborne segment and a ground segment. Two airborne system configurations were supplied, Hybrid I and Hybrid II. The former utilizes an airborne computer memory unit (CMU) to provide immediate diagnostic information to the pilot in-flight and to the maintenance crew post-flight. Additional detail is printed when data is processed at the ground segment. The airborne segment photo in Figure 1-2 depicts this Hybrid I configuration. In Hybrid II operation an airborne recorder is substituted for the CMU; and data processing is accomplished entirely by the ground segment. In both cases a hard copy printout of the results of diagnosis is provided at the ground segment.

The prototype system software consists of all the computer programs required for the AIDAPS systems. These include gas path, vibration and functional/mechanical logics, data validation, fault detection and annunciation functions. The airborne software resides in the computer memory unit (CMU). All software is used by the Diagnostic Analyzer (DA) since the DA performs the CMU functions when the system is operating in Hybrid II mode (with a recorder instead of a CMU). For the prototype application phase, AIDAPS hardware was fabricated and installed on UH-1H aircraft in preparation for the Phase V tests.

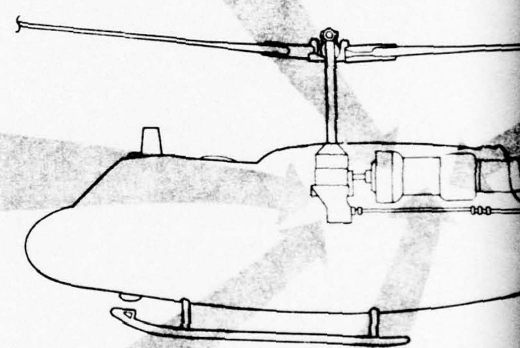
Phase V testing comprised two parts, a performance test period and a demonstration test period. During the performance test period, implant testing was done using the prototype hardware for the multiple purpose of system check-out and debugging and in order to refine the logic limits for this UH-1H test with the implant degradation levels being tested. A demonstration was given at Ft. Rucker at the end of this period. This marked the beginning of the demonstration test period in which 163 different degradation conditions were successfully tested. Some tests were unsuccessful due to various causes including equipment failures, improper test conditions, etc. A total of 120 of the degradations were annunciated automatically using the limits implemented in the prototype hardware. Photographs of a few of the implants tested are shown in Figure 1-3. These represent the levels of degradation which were considered "flyable" for this test and which were flown and detected. The following table identifies the parts depicted.

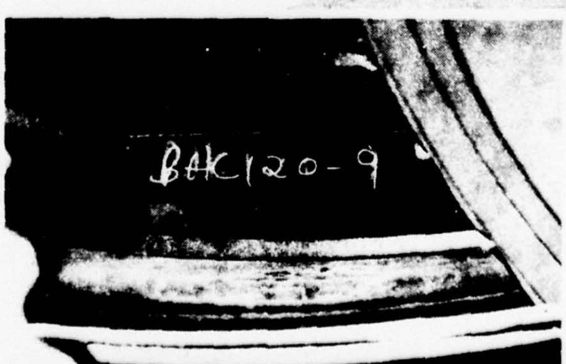
# AIDAPS - DEGRADED PART D


• COMPRESSOR



• BEARING

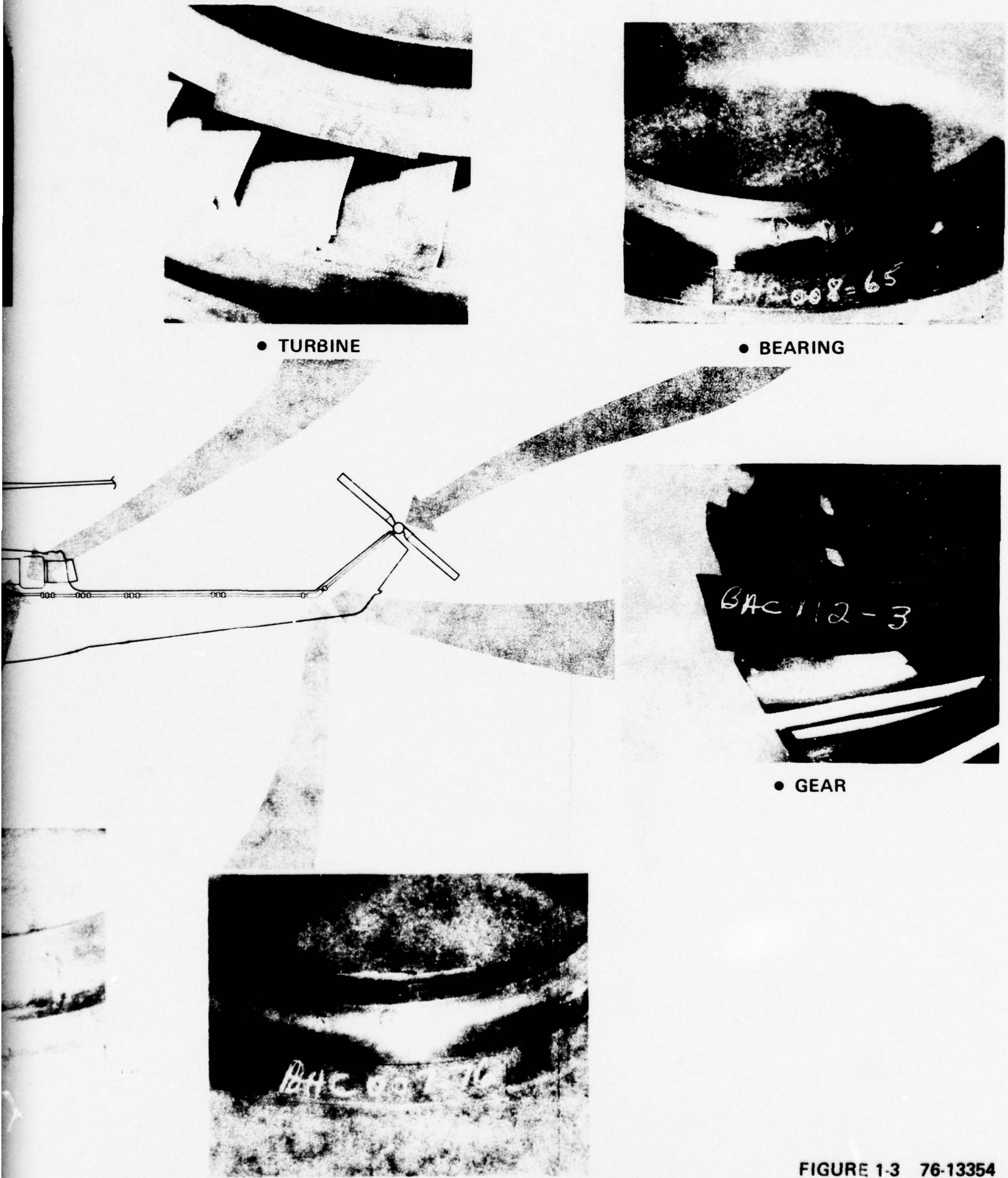





• GEAR

• BEARING

## RT DETECTION CAPABILITY

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#### Engine Implants

| | |
|---|---|
| AID 132/133 | Compressor rotor/stator |
| ATB 201 | Gas producer nozzle first stage |
| AID 145 | Bearing No. 4 |

#### Transmission Implants

| | |
|---|---|
| BHC 120 | Input drive quill-pinion gear |
| MAIC 018 | Input drive quill-triplex ball bearing |

#### $42^0$ Gearbox Implants

| | |
|---|---|
| BHC 007 | Duplex ball bearing |
| BHC 112 | Gear |

#### $90^0$ Gearbox Implant

| | |
|---|---|
| BHC 008 | Duplex ball bearing |

During the Phase V test period, the prototype hardware used to prove the feasibility of the AIDAPS concept was evaluated. Several hardware changes were incorporated and further evaluated to assess the improvement in system reliability. Based on this experience and additional recommendations regarding system performance improvements, projected system reliability of future system hardware can meet the reliability requirements of the AIDAPS system specification.

## 1.3 CONCLUSIONS AND RECOMMENDATIONS

In view of the demonstrated success of the AIDAPS hardware in detecting component part degradations automatically, and in view of the large savings which are projected for AIDAPS application to the AAH, it is recommended that steps be taken to implement AIDAPS on the AAH in a timely manner so that the full benefits of including AIDAPS in the first production run be realized.

To this end, certain system modifications require investigation and incorporation in the following areas:

- System hardware changes to improve system performance
- Maintenance message content and format changes to improve the maintenance personnel/system interface
- Incorporation of prognosis logic to enable prediction (and determination of) limiting part condition for removal from aircraft use.

## 2. PHASE 1 - DATA COLLECTION

### 2.1 TASK 1: HARDWARE DESIGN

This task required the design of (1) a gas path data collection system, (2) a vibration data collection system, and (3) an aircraft installation design. Initially, two gas-path systems and two vibration systems were installed on four Bell Helicopter Company UH-1H helicopters. Thus, gas path and vibration data could be collected on a non-interference basis. Later, the two helicopters equipped with gas path systems were equipped with vibration systems enabling gas path and vibration data to be collected simultaneously.

The gas-path systems were designed to accept input signals from sensors and wiring already existing on the two gas-path aircraft. The vibration systems were designed for installation on standard UH-1H helicopters, and the designs included all required aircraft wiring and sensor mounting brackets.

#### 2.1.1 Gas Path Data Collection System

The gas path data collection systems were first installed on two UH-1H helicopters, assigned the code names Bearcat II (BCII) and BC12. These systems included provisions for collecting mechanical/functional data. Later, these aircraft were fitted with vibration systems, in addition to the gas path system.

The gas path system was made up of a number of sensors located throughout the aircraft which measured various physical parameters, and input the electrical signals to the data collection system components, where they were conditioned and digitally recorded on magnetic tape. In addition to the sensors, the gas path data collection system was made up of the following components:

(a) A Signal Preconditioner (SPC) which was primarily used to precondition sensor signals. In addition, this box contained the switching for the +28v prime power to the system and contained tie points used for programming the FDAU.

(b) A Flight Data Acquisition Unit (FDAU) was used for additional signal conditioning of sensor signals, converting these conditioned analog signals to digital form and providing a properly formatted digital data stream to the magnetic tape recorder.

(c) A Digital Data Recorder (DDR) was used to record the AIDAPS data on a 7 track, 556 BPI, IBM-compatible digital tape.

(d) A Data Entry Panel (DEP) was used to input various documentary data on the flight tapes by means of manual switches. Prime power to the AIDAPS system was controlled by an on-off switch located on the DEP which in turn controlled a relay in the SPC to switch the +25-v aircraft power to the system.

(e) An inverter (Abbott P/N S12-115A-400) converted +28-vdc aircraft power to the 115 vac, 400-Hz power required for the system. A 15 amp circuit breaker had been installed between the +28-vdc aircraft supply and the AIDAPS system hardware.

NOTE: The FDAU's, DDR's, and DEP's described in subparagraphs (b), (c), and (d) above were AiResearch-owned equipment and were consigned to this program on a no-charge basis.

A more detailed description of the equipment design can be found in Reference 1, Section 2.1.1.

### 2.1.2 Vibration Data Collection System

Early in the program, two helicopters were configured for vibration data collection, BC13 and BC14. However, before any useful data collection flying was accomplished, BC13 experienced mechanical difficulties and was removed from service. The vibration data collection system from BC13 was removed and installed in BC12, one of the gas path aircraft, after relocating gas path equipment so both systems could be accommodated. Still later in the program, Contract Modification P00006 added two more vibration systems, which were installed in BC11 (along with gas path equipment), and in a new BC13.

The vibration data collection system consisted of: (1) 24 sensors located throughout the aircraft to monitor vibration levels of the engine, transmission, and drive trains; engine speed, torque, and hanger bearing temperature; (2) an analog signal preconditioner which received the sensor signals, conditioned these signals and output them to the airborne recorder: (3) a multichannel analog recorder which was designed to accept up to 14 channels of analog data at one time (in the AIDAPS system, 1 channel was reserved for recording of either a voice input or a flight mode indication); (4) a control panel located on the cockpit pedestal which controlled system power, turned the recorder on and off, provided an indication of recorder status, and contained a rotary switch for selection of mode or voice recording to the 14th channel of the analog recorder.

A more detailed description of the system components can be found in Reference 1, Section 2.1.2.

## 2.2 TASK 2: HARDWARE SYSTEM AND INSTALLATION

### 2.2.1 Hardware Fabrication

The AIDAPS hardware consisted of three categories of equipment, namely:

(a) Equipment specifically designed and fabricated for the AIDAPS program.

(b) Existing AiResearch-owned equipment which was consigned to the AIDAPS program.

(c) Equipment purchased for the AIDAPS program.

All sensors used for both the gas path and vibration data collection systems fell within category (c). All items in categories (a) and (c) were

considered deliverable items and Army-owned equipment. Items which fall in category (b) are AiResearch-owned equipment and were consigned to the AIDAPS program in the interest of cost saving on the program.

Reference 1, paragraph 2.2.1 lists the equipment as to whether it was consigned or deliverable hardware:

### 2.2.2 Hardware Installation

#### 2.2.2.1 Gas Path Data Collection System

The gas path system installation consisted of four "black boxes" which were shock-mounted on a plywood pallet in the cabin to the right of the transmission area. A data entry/control panel was located on the cockpit pedestal between the pilot's seats. The four "black boxes" consisted of two units used for signal conditioning and data formatting (SPC and FDAU), a digital recorder, and a power supply. Various sensors were located throughout the aircraft to measure physical parameters such as temperature, pressure, position and flow rates. Circuit breakers were added to protect the ±28 v and 115 v aircraft power sources to the AIDAPS system.

#### 2.2.2.2 Vibration Data Collection System

The vibration data collection system hardware consisted of two major "black boxes" shock-mounted on a plywood pallet in the cabin to the right of the transmission area. A data entry/control panel was located on the pedestal between the pilot's seats. The two major "black boxes" consisted of a 14-channel analog recorder (AR), and a signal preconditioner (SPC). Nineteen velocity pickup or accelerometer-type sensors were installed throughout the aircraft to measure and record vibration levels on the transmission, engine, and drive trains. A circuit breaker had been added to protect the +28-v aircraft power source to the AIDAPS system hardware.

#### 2.2.2.3 Installation Drawings

The Master Index List (IL 1100068), a Section Index List (IL 1101536), and referenced Data Lists (DL 1400102, 1400103, 1600293 and 1600294) list the drawings which describe both the gas path and vibration data system installations. These lists are contained in Appendix E of Reference 1.

Installation approval and flight clearance was received for the gas path aircraft, BC11 and BC12, in late October 1973. Installation approval and flight clearance was received for the vibration aircraft BC13 and BC14 in early January 1974.

## 2.3 TASK 3: DATA COLLECTION

Data Collection activities consisted of (1) engine test cell data collection from the Avco Lycoming Division (ALD) in Stratford, Conn., (2) test cell data for gearboxes and transmissions from Bell Helicopter Co. (BHC) in Fort Worth, Texas, and (3) flight testing at Cairns Field, Ft. Rucker, Alabama. The major part of data collection was in the flight testing activities at Fort Rucker.

### 2.3.1 Test Cell Data Collection

Most of this activity is discussed in Section 3 of this report, since the AiResearch work in this part of the program was primarily one of data reduction and analysis after receiving the collected data from LHC and ALD.

### 2.3.2 Flight Testing

Flight testing at Ft. Rucker was performed with four UH-1H helicopters instrumented with data collection hardware. Initially, two of the aircraft, designated as Bearcats 11 and 12, were fitted for collecting gas path and functional/mechanical data, and two more aircraft, Bearcats 13 and 14 were instrumented for vibration data collection. Due to mechanical difficulties with BC13, hcwever, it was removed from service before any useful data had been collected, and its vibration equipment was removed and installed in BC12, in addition to the gas path equipment, which had to be relocated so both systems could be accommodated.

Still later in the program, a Contract Modification added two more vibration systems; one system for a new BC13, and the other installed in BC11, along with the gas path equipment.

#### 2.3.2.1 General Procedures

Data collection procedures comprised scheduling of flights for baseline and implant data collection, methods of operating the data collection hardware, handling of recorded data tapes, documentation of the flights and the data reels, quick-look procedures and shipping of data to AiResearch for analysis.

Flight scheduling on a day-to-day basis was done at Ft. Rucker. An overall plan including the number of implants per month, identification of implants for the current month and the number of flights required for each implant was given in the Subsystem Test Plan and Revisions through No. 19, Reference 10.

During the entire Phase I data collection activities conducted at Fort Rucker, a total of 825 flights were flown, between January 1974 and August 1975. Table 2-1 gives a breakdown of these flights by aircraft number and type of test.

TABLE 2-1

PHASE I TEST FLIGHT SUMMARY

| Aircraft | Test Type | No. of Tests |
|---|---|---|
| Bearcat 11 | Baseline | 77 |
| | Implants | 109 |
| Bearcat 12 | Baseline | 61 |
| | Implants | 201 |
| Bearcat 13 | Baseline | 34 |
| | Implants | 53 |
| Bearcat 14 | Baseline | 77 |
| | Implants | 213 |
| | Total | 825 |

#### 2.3.2.2 Gas Path Data Collection

Two categories of flight profiles were used for gas path baseline data collection. The first category consisted of profiles which include the typical operating envelope for the UH-1H. These profiles were used for implant data collection as well as baseline generation. The second category consisted of flight profiles which were intended to provide data for the determination of the effects of other factors, such as altitude variations, on the spread of baseline data. These profiles were not intended for implant data collection. Scheduling of flights by profiles emphasized the first category in order to enlarge the statistical sample of data for typical flight regimes. Actual flight profiles used can be found in Section 2.3 of References 1 and 2.

Each flight was documented redundantly to reduce the possibility of lost data. A data entry form was filled out by AiResearch field personnel before each flight. This data form identifies run number, aircraft, engine, data, ambient conditions, flight profile and aircraft weight. Additionally, codes were provided to identify transducer changes and implant conditions. One copy of the hand-recorded data was sent to AiResearch, Torrance with the tape recordings. Reference 11 contains all of the flight test data generated during the program.

Two methods of gas path data collection system verification were provided. The first involved the use of a field test set. System operation on a channel-by-channel basis was checked using the field test set to verify that the system (exclusive of the recorder) was operating properly. Second, a quick look program was provided to facilitate aircraft turnaround during implant data collection. This program verified that the data collection system, including the recorder, was operating properly.

In order to have a quick turn-around inspection of the gas path data collected after flight testing, the quick-look type of data processing and analysis computer program was developed for operation at Ft. Rucker. Based on the printout information, the AiResearch field engineer could determine with a high level of confidence if pertinent and sufficient data had been recorded during the flight. Thus, he could release the engine or aircraft for the next scheduled flight, otherwise a reflight request was issued. The pilot could re-run the test flight on the same day if time allowed or he could schedule it for the next day.

The quick-look computer program is written in FORTRAN. The source program listing is given in Appendix B, Reference 1.

#### 2.3.2.3 Vibration Data Collection

Similar flight profiles were used for the vibration data collection, (See Section 2.3 of References 1 and 2) and the hand recorded data for vibration data collection was similar to that used for the gas path aircraft in that it included identification numbers, date, and ambient conditions. Additionally it included analog recorder track assignment and calibration information.

In flight, provision was made for the pilot to record ambient conditions, and flight leg identification number on the voice channel of the analog recorder.

The objectives of vibration tests performed consisted of the following:

(a) Aircraft Shakedown Tests

- To checkout, debug, troubleshoot the equipment function.
- To evaluate some sensor performance.
- To determine proper gain settings for each sensor channel and flight regime condition.
- To compare the frequency content of data with test cell data.

(b) Baseline and Evaluation Profile Tests

Baseline Data Collection Tests

- To establish baseline signatures and their statistical descriptions.

Evaluation Profile Tests

- To determine the influence and thereby to establish the relationship between certain parameters and the vibration signals flying several flight profiles

(c) Implant Tests

- To determine the signature deviation or creation due to implants.
- Implant order was 42 deg, 90 deg, hanger bearing, transmission and engine.
- 120 implants were planned.
- Followed same flight profile as for baseline data.

(d) Cell Tests - Conducted at BHC

- To provide basic component signature with and without implant.
- To permit analysis of aircraft-introduced signals.
- To provide prognostic data such as signature at certain initiating points and removal point and the corresponding limit levels.
- Test power settings and speed settings were in agreement with the power and speed settings employed in flight tests.

Additional details on vibration data collection, profiles, test log forms, etc., can be found in Section 2.3, References 1 and 2.

### 2.3.3 Status Reporting

A weekly status reporting system was established early in the program. This method identified the implants tested, profiles flown, and causes for aircraft downtime for each week of operation.

Section 2.3.2 of Reference 2 gives an example of this system, which allowed aircraft usage to be estimated reasonably accurate without the necessity of keeping an hourly log.

## 3. PHASE II - DATA REDUCTION AND ANALYSIS

The purpose of the data reduction and analysis phase was to convert the data base, collected during Phase I - Data Collection, into a useful informational form and then perform the analytical development of the AIDAPS fault detection logic using that information. The premise, or underlying theory for AIDAPS is as follows:

- Helicopter systems, subsystems, and components exhibit measurable performance and structural, mechanical, and functional characteristics during operation.
- Measurable characteristics change as a result of equipment deterioration, degradation, and maladjustments.
- Measurements and equipment condition are associative and correlative.
- Logic for association and correlation is definable and capable of automation.

The AIDAPS fault-detection logic was developed using three measurement categories to determine the deterioration of equipment condition. These measurement categories are:

- Vibration
- Engine gas path
- Mechanical/functional

This phase consisted of two major tasks. The first required the reduction and analysis of data to obtain the necessary information to determine the AIDAPS logic. The second was the task of logic determination. The salient features and accomplishments of these two tasks are presented in the following paragraphs.

### 3.1 TASK 1: DATA REDUCTION AND ANALYSIS

Experimental data were generated from three sources: (1) the test cell at BHC (gearboxes and transmission), (2) the test cell at ALD (engine), and (3) flight and ground runs at Ft. Rucker (aircraft). Certain data from the prior AVSCOM test-bed programs and subsidiary investigations by Parks College, St. Louis, also were utilized. The test cell data served two principal purposes: bases for parts/equipment safety-of-flight release, and precursor of aircraft fault detection ability. Flight and ground run data from the aircraft

at Ft. Rucker, provided the data base from which the ultimate detection character was determined and the appropriate diagnostic logic was established.

A system of procedures was established to handle the continuous flow of flight test data from Ft. Rucker. All data were computer-processed at AiResearch, Torrance, except for "quick-look" of Ft. Rucker gas path data performed on the IBM 360/50 computer at the Ft. Rucker computing center. In this fashion, timely results were made available to the flight test personnel. The procedures for processing this data are described in detail in References 1 and 2.

The details of the data reduction and analysis for gas path, mechanical/functional, and vibration categories are merely summarized in this section, but are covered in depth, supported by substantiating test data,in References 1, 2, 3, and 4.

### 3.1.1 Special Computer Software

Several computer programs were developed for handling the data reduction and analysis of the gas path and vibration data collected during Phase I of the program. These included:

(a) Gas Path Programs

- Quick-look data processing program
- Gas-path data processing and engineering units conversion program
- Gas-path data analysis program
- Gas-path sensor calibration history program

(b) Vibration Programs

- Vibration data processing and reduction program
- Broad-band amplitude analysis program
- Likelihood analysis program

Throughout the AIDAPS program, the earlier computer software was continually modified, as a result of the data collected and analyzed; or revised, as a consequence of the logic development. The initial listings for the gas-path quick look, gas-path data extraction/reduction, and vibration likelihood analysis programs, respectively, can be seen in Appendixes B, C, and D of Reference 1.

Some of the more significant changes that occurred in the software included (1) the addition of a Broad-Band Amplitude Analysis program to the Likelihood Analysis program in order to keep the number of bands down to a manageable level, (2) the storage of sensor calibration data in the gas-path

reduction program disk, (3) re-writing the vibration data processing and reduction program for the Digital Scientific META-4 Computer, from the IBM-1800. References 2, 3, and 4, Section 3.1.1 discuss these changes.

The finalized updated versions of the special computer programs developed for AIDAPS can be found in the seven documents covering Phase II software in Reference 12.

### 3.1.2 Gas Path Data Reduction and Analysis

#### 3.1.2.1 Data Sources

##### 3.1.2.1.1 ALD Test Cell Data

The objectives in this area of the program were to establish signatures of components with known grades of severity and under controlled conditions, and to document the performance of the diagnostic techniques. The Lycoming Division of AVCO (ALD) provided data collected from baseline and degraded engines operated in test cells of their Stratford, Connecticut facility.

This test cell activity offered an advantage of allowing an accurate measurement of referred air flow (W2C2). Referred air flow was calculated from inlet bellmouth measurements and an air flow calibration for the bellmouth. The change in this parameter is one of the independent variable outputs of the thermodynamic model. Moreover, measurement of this parameter enabled direct calculation of gas producer nozzle area, another output of the thermodynamic model, as well as several other parameters of interest.

Four of the outputs of the thermodynamic model could be calculated directly. Thus, these outputs could be correlated with the implants, and compared with independently calculated values of those quantities. For the remainder of the model output, such direct comparisons were not possible, since it was impossible to directly calculate gas producer turbine and power turbine efficiencies and power turbine inlet nozzle area without making significant assumptions that would strongly influence the comparisons. (In effect, the value of one of those three parameters must be assumed in order to calculate the other two.)

It was possible, however, to directly compute the combined efficiency of the two turbines. Since this efficiency is approximately equal to the sum of the turbine efficiencies weighted by their contributions to the total power outputs of the turbines, it was possible to check the consistency of the model-calculated turbine efficiencies.

The directly calculated performance parameter changes showed excellent agreement with the changes calculated by the AIDAPS T53 thermodynamic model. (Includes the following: compressor efficiency and airflow, turbine inlet temperature, and gas producer turbine nozzle area.)

##### 3.1.2.1.2 Ft. Rucker Flight Test Data

The USAAVNTBD provided test cell, METS acceptance-type test and flight test data for UH-1H engines tested at Ft. Rucker, Alabama. The test cell and

METS data was analyzed for its possible correlative value and because it reflected ground test stand instrumentation capabilities that could be considered for AIDAPS for certain engine-airframe combinations.

Flight test data, collected on a number of engine/aircraft combinations, served as a data base for subsequent logic determination tasks and evaluation of implementation alternatives. Flight test data were collected using engines in baseline condition and with degraded parts that had been tested in ALD test cells so that a comparison of cell test and flight test data would reveal the affect of flight conditions on the output of the thermodynamic model of the engine.

In comparing the alternatives for use as independent variable ($N_1'$ vs $P_3'$), it was found that VIGV operation favored the use of $P_3'$; however, flight test results showed better $N_1'$ repeatability due, at least in part, to problems with the compressor inlet and discharge pressure transducers. This led to a selection of pressure transducers with improved performance for the prototype system.

3.1.2.2 Flight Data Analysis Results

Flight testing included engine baseline operation investigation and implant tests. The baseline tests were designed to uncover any unforeseen profile or configuration effects such as variable back pressure effects that might require modifying initial gas path monitoring logic. No such effects were found significant.

The baseline tests also were used to establish parameter measurement and computational repeatability. Measured parameter repeatability for a series of flights covering the complete range of flight profile specification parameters is shown in Figure 3-1. These parameters included three ranges of gross weight, three altitude ranges, three airspeed ranges, and naturally occurring ambient temperature variations. None of the above variables correlated with the variations in the referred gas path variables (examined both for intra- and interflight variations). The dashed lines in the figure are not limits, but are essentially figures of merit. Variations that remain within the dashed lines indicate excellent repeatability. Typically, the goodness of parameters with respect to repeatability has been (in order of decreasing repeatability) $T_9'$ (EGT), $T_3'$, $P_3'$, $W_f'$, and SHP.

Repeatability is partly dependent on the degree of engine stabilization (steady-state). In general, it has been found that 30 sec provides adequate stabilization following a throttle transient and that EGT is the best indicator of stabilization. The approach used to specify stabilization provides a built-in, adaptive stabilization period that adjusts the waiting period on the basis of the magnitude of the transient. Originally, 7-min periods were scheduled for each altitude-airspeed condition in a test flight, resulting in 5 to 6 min of stabilized engine operation following a simple power change (no altitude change) and a little less for changes in altitude as well as power. The 7-min periods were shortened to 5 min. In the operational case, whenever

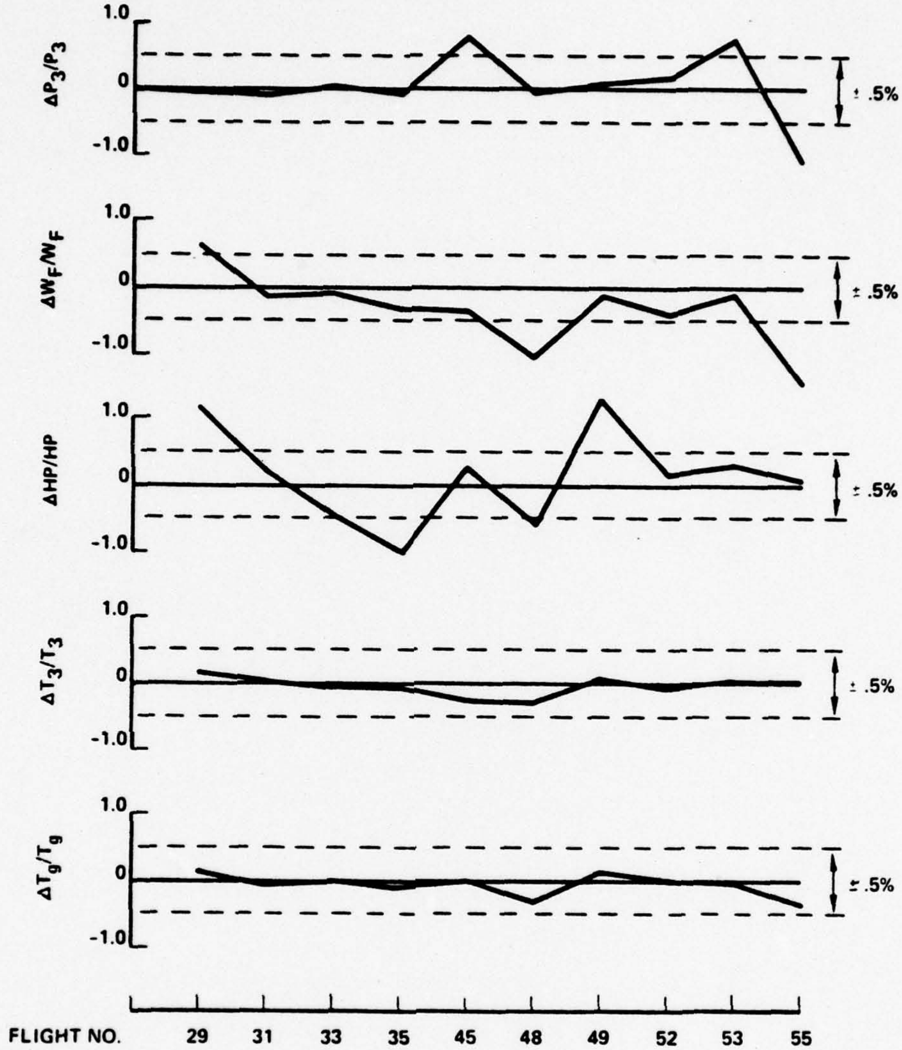



Figure 3-1. Flight-to-Flight Scatter - Measured Parameters

the collective is not moved for 1 to 2 min, stabilization sufficient for gas path monitoring will generally result (where sufficient refers to time duration, not quality; stabilization by itself refers to quality).

Figure 3-2 shows the model results obtained from the measured variations shown in Figure 3-1. The dashed lines denote standards rather than limits. Flight-to-flight scatter and within-flight scatter indicate adequate measurement repeatability for AIDAPS gas path monitoring.

Certain data collection system sensors, however, did not achieve the accuracy and repeatability required for the prototype system. Specifically, improvements were required in measurements of compressor inlet pressure, compressor discharge pressure ($P_3$) and torque pressure (SHP). New sensors and/or sensor installation locations were chosen for the prototype system implementation. Tests on these instrumentation changes during Phase I showed a significant improvement in data quality.

Thirty flights of data with the new transducers were analyzed, and in addition, a flight profile that was specially selected to reveal any problems in the overall gas-path system was used. This flight profile was flown at an altitude of 1000 ft with an air speed varying between 70 and 90 knots. The approximate duration was 1-1/2 hr. The profile enabled the acquisition of data for the same referred $N_1'$ at various time intervals. Pertinent data is given in Figures 3-3 through 3-5. Figure 3-3 shows the variability of the measured gas-path parameters as a function of time at essentially fixed gas generator referred speed. The variations with the exception of VIGV lie within the acceptable band of ±1/2 percent. The repeatability of the data for the same speed and for points corresponding at the beginning and end of the flight (over one hour) was found to be excellent. The variations for constant $P_3'$ (Figure 3-4) are smaller.

The model results corresponding to the variations of Figure 3-3 are shown in Figure 3-5. The variability of the data is within the expected accuracy of gas-path analysis.

#### 3.1.2.3 Review of Other Gas Path Monitoring Approaches

Three different approaches to gas path monitoring were investigated during Phase II. Among these were a pattern recognition approach, a computational approach, and the HIT approach currently in use by the U.S. Army. These approaches were reviewed to ensure development of an optimum approach for AIDAPS. For example, the notion of a quantitative index to the quality of diagnostic results (an estimated confidence level) as implemented in the pattern recognition approach (developed by R. Lazarick of NAPTC) appeared to be an excellent idea and was applied to AIDAPS, even though the basic pattern recognition approach was not sufficiently developed to be considered in a competitive sense. The approach being developed for AIDAPS still appears unsurpassed in combining wide diagnostic coverage with versatility and implementation ease. A description of these other approaches is given in Appendix D of Reference 2.

The HIT approach is a simple, manual approach for trending engine performance changes. The approach was analyzed and found to be a fairly effective means of monitoring hot section (turbine) condition, but it was also determined to be relatively insensitive to compressor degradation. Discussion with field unit personnel supports the latter conclusion.

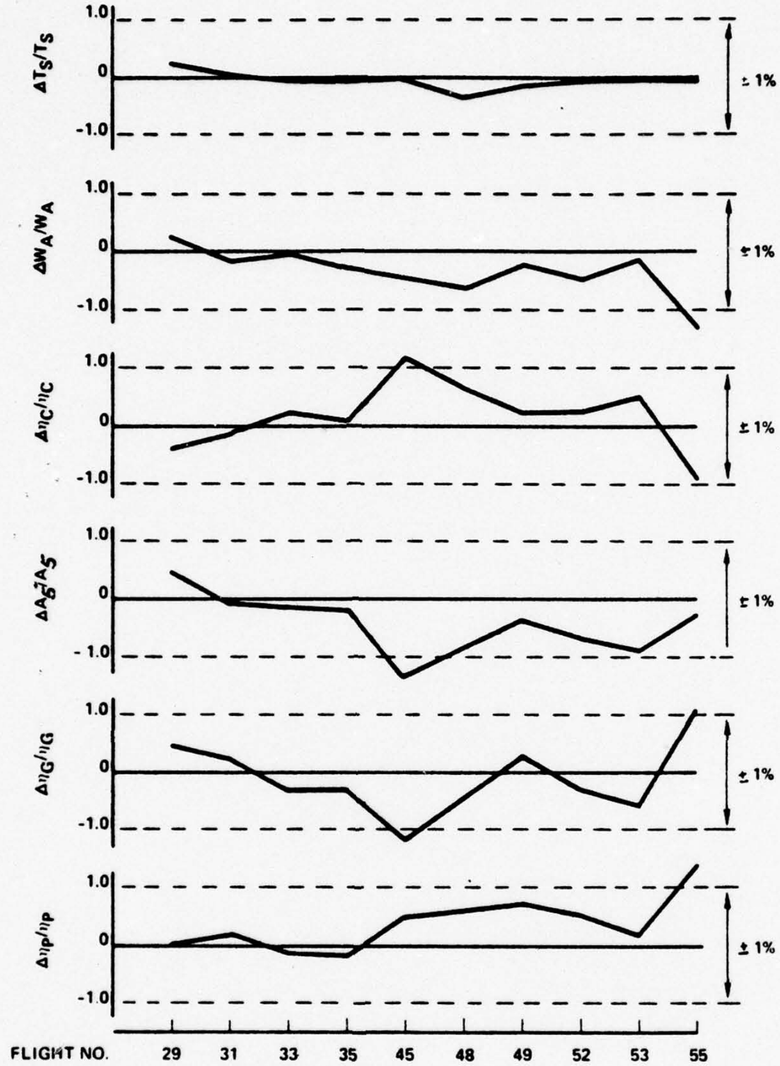



Figure 3-2. Flight-to-Flight Scatter - Calculated Parameters

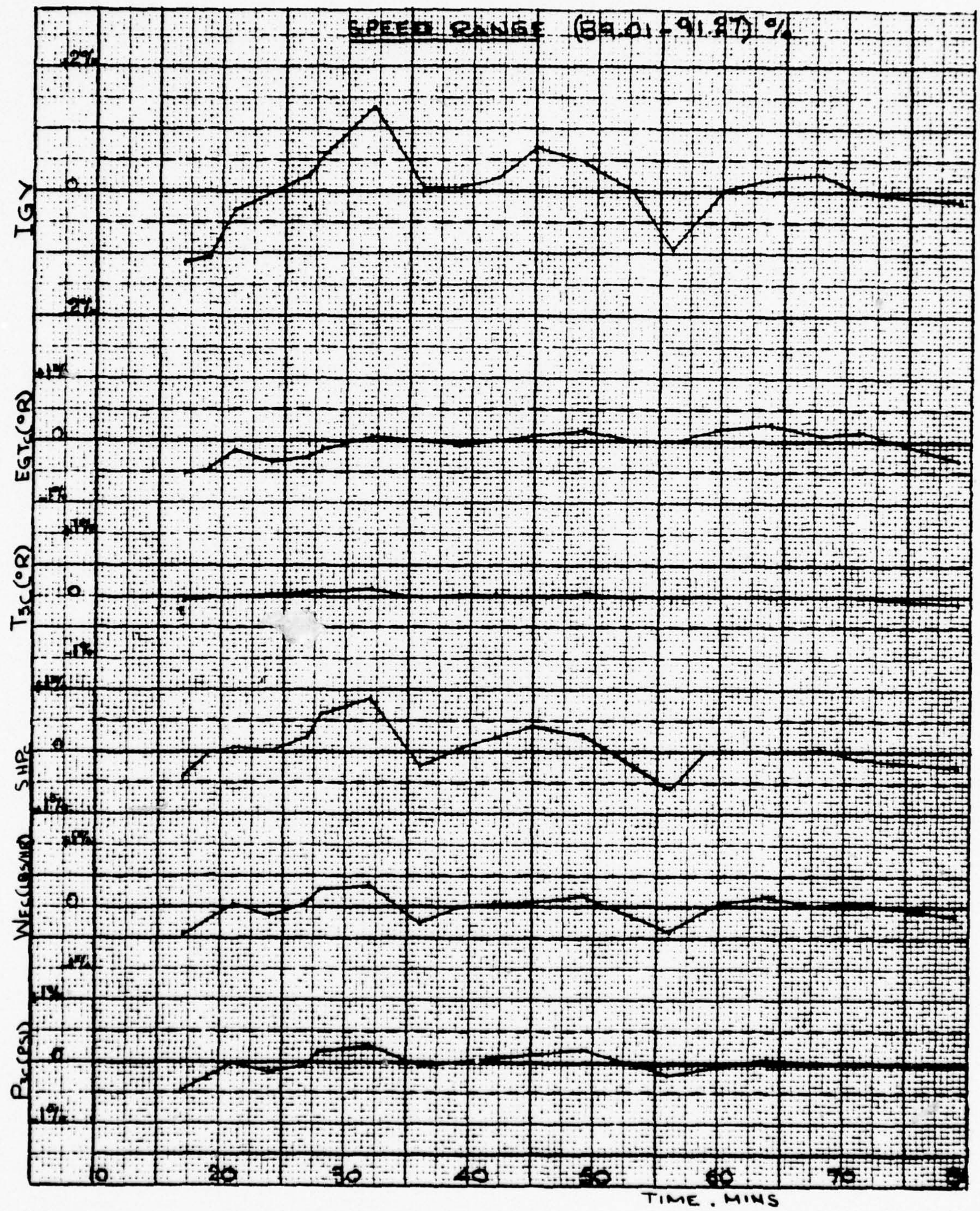



Figure 3-3. Gas Path Measured Parameter Variations as a Function of Corrected Gas Generator Speed

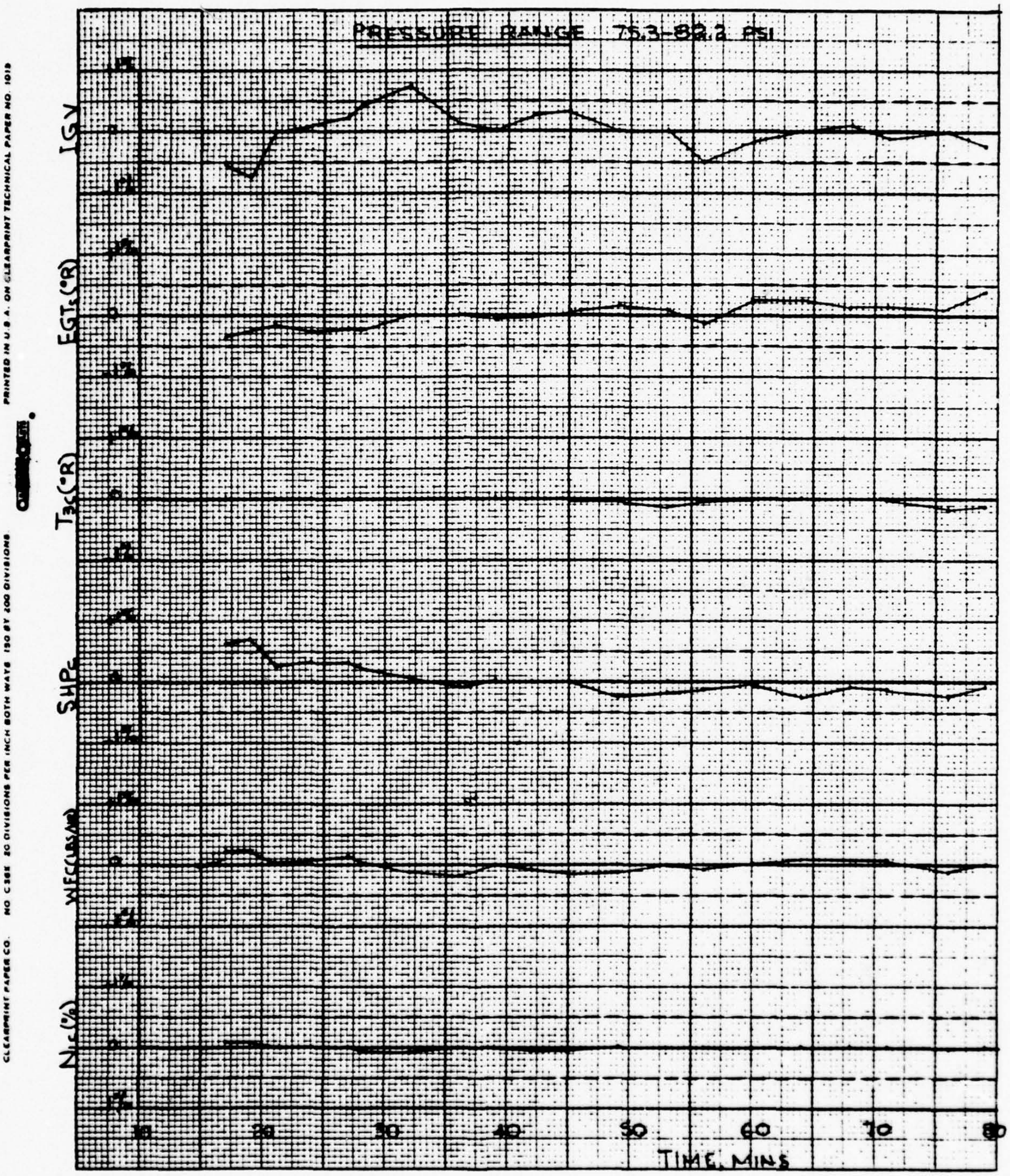



Figure 3-4. Gas Path Measured Parameter Variations as a Function of Corrected Compressor Exit Pressure

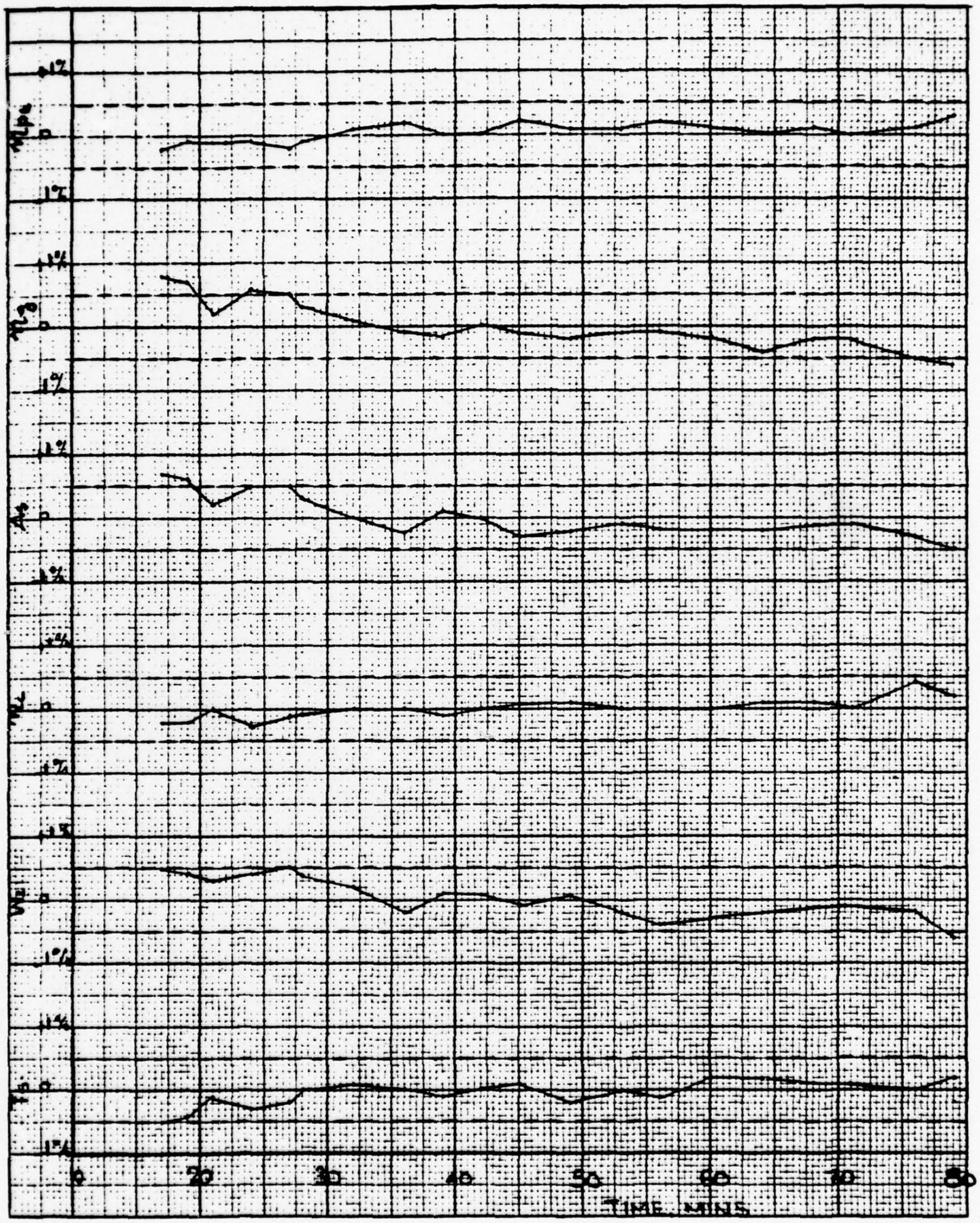



Figure 3-5. Gas Path Performance Parameter Variations

### 3.1.3 Vibration Data Reduction and Analysis

#### 3.1.3.1 Data Sources

##### 3.1.3.1.1 UH-1H Component Shaft Frequency Data

Basic shaft rotational frequency data for UH-1H components was supplied by BHC and ALD. Based on that data, more universal and extensive frequency tables for UH-1H engine, transmission, and gearboxes were established. Instead of tabulating the true vibration frequencies in Hz and listing them discretely at several shaft speed levels as was done previously, the vibration frequencies were normalized to either $N_1$, $N_2$, or $N_T$ (transmission input speed) depending upon whether the component is driven by $N_1$, $N_2$, or $N_T$. The use of ratios allowed the establishment of frequency relationships for all shaft speeds. Tables of component shaft frequencies are presented in Reference 2.

##### 3.1.3.1.2 Test Bed Program Data

Test bed program vibration data was not used extensively on the AIDAPS program due to its limited frequency range of coverage and the limited number of implants of large severity of degradation. Some highly degraded Phase E implant data was reduced and analyzed from both test cell and flight test portions of the program.

Analysis of the Test Bed Phase E test cell data indicated that an implant of the type tested can be easily detected from more than one frequency band and from more than one transducer location, consistently, with a quite large detection margin.

Analysis of the Phase E flight test transmission data indicated that some of the discernible transmission frequencies observed in the test cell data appear to show up in the flight test data, providing confidence that cell test findings can be used to anticipate flight test results.

##### 3.1.3.1.3 BHC Drive Train Test Cell Data

The purpose of the BHC test cell tests was to provide vibration signatures of baseline power train components (transmission and gear boxes) and components with degraded parts implanted. Initial tests on 42-deg gear boxes indicated that gear-box-to-gear-box vibration signature differences were significant enough to require individual baselines for every 42-deg gear box. Tests also indicated that although the shape of the implant vibration spectrum appears similar for different gear boxes, the response in unique bands varies, and will require detection logic which covers a broad range of frequency bands.

As cell testing proceeded, the usefulness of test cell data, in terms of (1) being a representative simulation of flight data, and (2) the timeliness of data for scheduling purposes, was questionable. Therefore, BHC cell tests to provide diagnostic data were curtailed in favor of flight testing, and the component test effort shifted to Ft. Rucker, Alabama.

#### 3.1.3.1.4 ALD Engine Test Cell Data

The purpose of the ALD test cell tests was to provide vibration signatures of baseline engines and engines with degraded parts implanted. Tests were conducted using degraded bearings and gears and unbalanced turbine rotors. Other tests determined the engine-to-engine variation in vibration signature.

Generally speaking, the degree of variation of vibration signature from one engine to another was a function of the sensor location; some locations varied more and some less, and the higher frequency portion of the spectrum varied more than the lower portion of the same spectrum. The ALD cell tests provided a basis for sensor location on the aircraft and also provided parts for implant flight tests at Ft. Rucker, Alabama.

#### 3.1.3.1.5 Ft. Rucker Flight Test Data

Flight tests were conducted at Ft. Rucker, Alabama, and baseline/implant vibration signature data provided on various combinations of aircraft-component-degraded part for engines, transmissions and gear boxes. The collected data formed the most significant part of the data base needed to develop the AIDAPS logic. In order to obtain a large sample of potential vibration signatures in the data base, the following numbers of components and parts were tested on four UH-1H helicopters..

| No. of Aircraft Components | | Parts Locations | Parts |
|---|---|---|---|
| Engine | 9 | 8 | 18 |
| Transmission | 6 | 15 | 31 |
| $42^0$ Gear Box | 7 | 6 | 23 |
| $90^0$ Gear Box | 7 | 6 | 28 |

These parts were tested under various conditions of aircraft weight and flight profile. Analysis of this data allowed determination of the affect of flight conditions such as altitude, airspeed, power level, and engine rpm, on baseline and implant repeatability.

### 3.1.3.2 Flight Data Analysis Results

#### 3.1.3.2.1 Baseline Data Repeatability

Data were collected from the same gearbox on the same aircraft over a one-week period. The gearbox was in baseline condition with no known faults implanted. All other flight conditions were kept constant during these tests. The processed results in PSD form indicated the features of the vibration signals were repeatable in each test sequence, and the quantitative amplitudes of the spectral frequencies were also within reasonable tolerances. This particular evaluation revealed that a gearbox has desirable long-term vibration signal stationary characteristics.

3.1.3.2.2 Implant Data Repeatability

This evaluation analysis was performed to determine if the long-term stationarity of the vibration signal can be abrogated by introducing faults into the mechanical system. Data were taken from tests using implant BHC 007, which is a 143 bearing with outer race spall. Results indicated the long-term stationarity of the data is still preserved.

3.1.3.2.3 Effect of Altitude

The influence of altitude on vibration signals was evaluated. Data at different altitudes, but with other conditions remaining the same, were compared. The results indicated the amount of variation of the vibration PSD appears to be of the same magnitude as existed in the constant-altitude case. As a conclusion, the altitude effect on the vibration signals from a gearbox appears to be negligible.

3.1.3.2.4 Effect of Shaft Rpm

The effect of shaft rpm on vibration signature was evaluated at a constant altitude and over a typical flight $N_1$ rpm range. $N_2$ was kept at 6600 rpm, as is practiced on the majority of the actual flights. Resultant PSD plots indicated that no significant change due to an rpm change from 90.5 percent to 93.5 percent is noticeable. This indicated that the rpm effect in the regular operating range in flight is negligible, possibly because of the extremely narrow operating range of $N_1$ rpm of a turboshaft aircraft (such as less than 5 percent). However, comparison of the data with ground idle showed significant changes in vibration amplitudes. Therefore, vibration monitoring applied to ground runs down to the ground idle point required rpm tracking and amplitude compensation to equalize the data to the same level as in the flight case.

3.1.3.2.5 Effect of Power Level Change

The 42-deg gearbox input quill accelerometer data from the tie-down tests were used for this evaluation purpose, since at tie-down, the power levels to the tail rotor drive at the test steps are known. Similar to the $N_1$ rpm effect, the influence of operating power level appears to have more significance at the low range (i.e., from idle to 50 hp). At the higher power range (i.e., from 50 hp to 100 hp), this influence becomes saturated and has no influence.

3.1.3.2.6 Effect of LRU Serial Number and/or Aircraft Serial Number

This analysis was performed to evaluate the vibration signature differences between gearboxes and between aircraft. Three different gearboxes (i.e., $F_1$, $F_5$, and $F_6$ were installed on two aircraft (Bearcats 12 and 14) and tested under the same conditions. The gearbox-to-gearbox difference in vibration PSD signature was observable in PSD plots. The combined effect of different gearboxes on different aircraft appears to be even more significant. The analysis results indicate the need for individual baselines.

#### 3.1.3.3 Review of Other Vibration Analysis Techniques

A review and evaluation of ten vibration analysis techniques was conducted (See Reference 13). Sixteen different criteria were used to evaluate the discrimination capabilities, implementation requirements, and state of development of each technique. Each criteria was assigned a weight and a total score generated for each technique. A ranking with a discussion of relative strength or weakness provided a rationale for selecting candidate techniques for further development.

The analysis showed that the Likelihood Index analysis technique being used to develop the AIDAPS was superior to the other techniques considered, and still offered the best probability of meeting the Army's goals for a useful diagnostic system.

### 3.1.4 Functional/Mechanical Data Reduction and Analysis

#### 3.1.4.1 Data Sources and Analyses

The functional/mechanical fault detection logic for the engine and aircraft was developed from recommended logic and supporting design information supplied by BHC and ALD. The basis for the recommended logic was the ability to detect faults in those aircraft/engine components which have high failure rates, high maintenance costs, and/or high safety of flight criticality. This logic was reviewed by AiResearch, together with engine test data obtained from cell tests at ALD in Stratford, Connecticut, and information obtained from the operator's and maintenance manuals for the Army Model UH-1 D/H Helicopter (TM55-1520-210-10, TM55-1520-210-20, and TM55-1520-210-34). This review also included recommendations of maintenance personnel of AVSCOM and TRADOC. The results of these analyses are reported in Reference 3 and lead to the development of the functional/mechanical logic.

## 3.2 TASK 2: LOGIC DETERMINATION

### 3.2.1 Gas Path Logic

#### 3.2.1.1 T53 Model Development and Application

The measurable properties (pressures, temperatures, etc.) of an engine's internal flow are determined by a set of parameters made up of boundary conditions and component performance parameters. Boundary conditions include all external variables that influence the internal flow, such as engine inlet conditions, bleed loads, accessory horsepower extraction, variable geometry, and so on. Also included is a variable that can be referenced to the engine operating point (energy per second input by the fuel, for example, could be chosen). The component performance parameters include all the internal engine component characteristics such as efficiencies and areas that affect the internal flow properties. All of the latter characteristics are linked to the physical condition of the components. The internal flow properties may be characterized as dependent variables, and the boundary conditions and performance parameters may be considered as independent variables.

If an engine's internal flow properties are from time to time measured at the same boundary conditions, then any changes observed in those properties must be due to changes in the engine component performance characteristics. This fundamental relationship between engine condition and engine variable measurements is the foundation for all engine performance monitoring approaches. The cataloging of engine variable measurements for various sets of boundary conditions (say, at installation of the engine) forms the notion of engine baselines.

The set of relationships, or equations, describing the gas turbine engine process and linking the flow properties to the component performance characteristics, constitutes an algebraic thermodynamic model of the engine.

Equations constituting this model for the T53 engine are listed in Table 3-1. Equation 8 is applicable to operation of the independently controlled power turbine speed at optimum (the speed at which maximum turbine efficiency is obtained for the gas generator operating points). This constraint is satisfied in application by introducing a correction that converts the actual SHP of the power turbine to that which would be obtained at optimum.

Equations 3 and 6, without the nozzle flow coefficients $K_5$ and $K_7$, are not perfectly valid applications of the nozzle flow function. In differential form, however, they adequately represent the relationships between the dependent and independent variables.

Selection of dependent variables is dependent upon measurability, availability of measurement location (e.g., pressure taps), already-available measurements, and in general, cost-effectiveness. Selections for the T53 in corrected form were:

$P4C = P4/\delta$ corrected compressor discharge pressure

$T3C = T3/\theta$ corrected compressor discharge temperature

$WFC = W_f/(\delta\theta^{.712})$ corrected fuel flow

$SHPC = Q(N_2)/(80\delta\theta^{.587})$ corrected shaft horsepower

$EGTC = EGT/\theta$ corrected exhaust gas temperature

Note that the corrected variables encompass the inlet pressure and temperature boundary value parameters since they are all referred to inlet conditions through the variables $\delta = P_{t2}/14.7$ psi and $\theta = T_{t2}/518.7\ ^{o}R$.

Selection of independent variables is dependent on the relation of these variables to significant engine degradation modes and on the aforementioned selection of dependent variables. Selections for the T53 were:

$T5C = T_{t5}/\theta$ corrected GP turbine inlet temperature

$W2C = W_2\sqrt{\theta}/\delta$ corrected compressor airflow

TABLE 3-1

ALGEBRAIC THERMODYNAMIC MODEL

1. Compressor Efficiency

$$\eta_c = \frac{\left(\frac{P_{t3}}{P_{t2}}\right)^{\frac{\gamma_c - 1}{\gamma_c}} - 1}{\frac{T_{t3}}{T_{t2}} - 1}$$

2. Combustor Heat Balance

$$\eta_B w_f (HV) = w_3 (T_{t5} - T_{t3}) C_{p,air} + w_f (T_{t5} - T_f) C_{p,fuel}$$

3. Gas Producer Turbine Inlet Nozzle Flow

$$\frac{w_5 (T_{t5})^{1/2}}{P_{t5} A_5} = K_5 \left( \frac{\gamma_5 g}{R} \left( \frac{2}{\gamma_5 + 1} \right)^{\frac{\gamma_5 + 1}{\gamma_5 - 1}} \right)^{1/2} \quad \text{(choked)}$$

4. Gas Producer Turbine Efficiency

$$\eta_G = \frac{1 - \frac{T_{t7}}{T_{t5}}}{1 - \left(\frac{P_{t7}}{P_{t5}}\right)^{\frac{\gamma_G - 1}{\gamma_G}}}$$

5. Compressor - G-P Turbine Work Balance

$$w_2 \Delta H_c + HP_{EXT} \cdot 550/J = \eta_m w_5 \Delta H_G$$

5A. $$\Delta H_c = C_{p_c} \frac{T_{t2}}{\eta_c} \left[ \left(\frac{P_{t3}}{P_{t2}}\right)^{\frac{\gamma_c - 1}{\gamma_c}} - 1 \right] = C_{p_c} (T_{t3} - T_{t2})$$

5B. $$\Delta H_G = C_{p_G} T_{t5} \eta_G \left[ 1 - \left(\frac{P_{t7}}{P_{t5}}\right)^{\frac{\gamma_G - 1}{\gamma_G}} \right] = C_{p_G} (T_{t5} - T_{t7})$$

6. Power Turbine Inlet Nozzle Flow

$$\frac{W_7 (T_{t7})^{1/2}}{A_7 P_{t7}} = K_7 \left\{ \frac{2 g \gamma_7}{R(\gamma_7 - 1)} \left[ \left(\frac{P_{t9}}{P_{t7}}\right)^{\frac{2}{\gamma_7}} - \left(\frac{P_{t9}}{P_{t7}}\right)^{\frac{\gamma_7 + 1}{\gamma_7}} \right] \right\}^{1/2}$$

7. Power Turbine Efficiency

$$\eta_p = \frac{1 - \frac{T_{t9}}{T_{t7}}}{1 - \left(\frac{P_{t9}}{P_{t7}}\right)^{\frac{\gamma_p - 1}{\gamma_p}}}$$

8. Shaft Horsepower Output

$$SHP^* = J W_7 C_{p_p} T_{t7} \eta_p \left[ 1 - \left(\frac{P_{t9}}{P_{t7}}\right)^{\frac{\gamma_7 - 1}{\gamma_7}} \right] \eta_{RG}$$

*optimum (optimum $N_2$ schedule)

($\eta_{RG}$ = reduction gearbox efficiency)

$\eta_c$ = Compressor efficiency

$A_5$ = GP turbine inlet nozzle area

$\eta_G$ = GP turbine efficiency

$\eta_p$ = Power turbine efficiency

$A_7$ = Power turbine inlet nozzle area

The above set of seven variables was reduced to six by assuming that $D(A_7) = KD(\eta_p)$.

The equations constituting the differential version of the thermodynamic model are listed in Table 3-2. The equations have been slightly simplified and are in non-reduced form; that is, equations 7 and 8 can be incorporated into the first 6 equations through substitution, and D(P4C) can be similarly eliminated from the right-hand side of the equation where it appears.

A more thorough treatment of the development of the gas-path thermodynamic model is presented in Appendix A of Reference 6.

The basic T53 engine gas path monitoring approach was proven valid and workable by application to data obtained from test cell and flight tests of T53 engines with degraded gas path components installed. Flight test data also demonstrated that data acquired under operational conditions are sufficient for this type of analysis. Engine operation is sufficiently stable and factors such as configuration (gross weight), altitude, airspeed, and pilot technique or mission type have no direct influence on the approach (i.e., no alteration of basic logic required), except that in establishing engine baselines (in the operational case), it would be analytically convenient and expedient to have a special profile flown to ensure broad coverage of the engine power range.

The basic philosophy underlying the gas path logic is that gas path component degradation is a relatively slow-rate process, and therefore warrants first attention by maintenance personnel, rather than the pilot. A notable exception, of course, is FOD, which is always potentially serious but considered more readily detected by vibration. No safety-of-flight outputs are therefore recommended to result directly from the onboard portion of the gas path analysis.

Functional checks performed during flight include max.-limit checks on $N_1$ and EGT. Exceedance of these limits can reflect serious gas path condition. Thus, gas path degradation is assumed to affect flight safety or mission completion only to the extent that it prohibits performance of mission functions within the operational limits placed on engine parameters ($N_1$ and EGT).

The structure of the analysis logic is as follows (for Hybrid I). Individual engine baselines are stored in the airborne unit. During stabilized

TABLE 3-2

DIFFERENTIAL THERMODYNAMIC MODEL
(SIMPLIFIED, UNREDUCED)

1. $D(P4C) = 0.5\ D(T5C) + D(W2C) - D(A5)$

2. $D(T3C) = -(1 - T_{t2}/T_{t3})\, D(\eta_c) + \beta_c\, (1 - T_{t2}/T_{t3})\, D(P4C)$

3. $D(WFC) = (1/(1 - T_{t3}/T_{t5}))\, D(T5C) + D(W2C) + \dfrac{(1 - T_{t2}/T_{t3})}{(T_{t5}/T_{t3} - 1)} \Big[ D(\eta_c) - \beta_c\, D(P4C) \Big]$

4. $D(SHPC) = D(W2C) + D(\eta_p) + D(T_{t7}) + \beta_p\, D(P_{t7})$

5. $D(EGTC) = -(T_{t7}/T_{t9} - 1)\, (D(\eta_p) + \beta_p\, D(P_{t7})) + D(T_{t7})$

6. $D(W2C) + 0.5\ D(T_{t7}) - (1 + \beta_7)\, D(P_{t7}) - D(\eta_p) = 0$

7. $D(P_{t7}) = (0.5 + 1/\beta_G)\, D(T5C) + D(W2C) + (1/\beta_G)\, D(\eta_c) - D(A_5) + (1/\beta_G)\, D(\eta_G) - (\beta_c/\beta_G)\, D(P4C)$

8. $D(T_{t7}) = (T_{t5}/T_{t7})\, D(T5C) + (T_{t5}/T_{t7} - 1) \Big[ D(\eta_c) - \beta_c\, D(P4C) \Big]$

$$\beta_c = \frac{\gamma_c - 1}{\gamma_c} \Big/ \left(1 - (P_{t3}/P_{t2})^{\frac{1-\gamma_c}{\gamma_c}}\right) \qquad \beta_G = \frac{\gamma_G - 1}{\gamma_G} \Big/ \left((P_{t7}/P_{t5})^{\frac{1-\gamma_G}{\gamma_G}} - 1\right)$$

$$\beta_p = \frac{\gamma_p - 1}{\gamma_p} \Big/ \left((P_{t9}/P_{t7})^{\frac{1-\gamma_p}{\gamma_p}} - 1\right) \qquad \beta_7 = \frac{\beta_p}{(\gamma - 1)} \left[ \frac{\gamma + 1}{2} - \left(\frac{P_{t7}}{P_{t9}}\right)^{\frac{\gamma - 1}{\gamma}} \right]$$

ASSUMPTIONS: $D(A7) = -D(\eta_p)$, $D(P3C) = D(P4C) = D(P5C)$,
$D(\eta_B) = D(W_L) = D(\eta_m) = D(HP_{EXT}) = D(P9C) = 0$

Note: $D(x) \equiv d(\ln(x)) = \dfrac{dx}{x}$

engine operation, the referred engine variables are computed and compared to their baselines. The resulting deltas (percentage difference from baseline) are compared to two sets of limits, one set denoting sensor malfunction (large, incredible exceedances), the other set denoting possible engine degradation. In the event of an exceedance, a sensor error or engine degradation signal is output, as appropriate. Irrespective of whether a limit is exceeded, the data are compacted and stored, resulting in 10 to 20 data sets per flight, where a data set consists of one value of each parameter monitored. In the event of an engine degradation output, the stored data are transferred to the ground unit where the diagnostic processing is performed and maintenance action messages generated.

### 3.2.1.2 Selection of Independent Variables ($N_1$ vs $P_3$)

In detecting engine performance degradation, measured referred engine variables are compared to their baseline values at the same value of an independent variable, which is also measured. Two variables, $N_1/\sqrt{\theta}$ and $P_3/\delta$ ($N_1'$ and $P_3'$), were considered for this purpose to determine which yielded the best results both from practical and theoretical aspects. The words "$N_1$ model" and "$P_3$ model" have been used to identify the parallel usage of the two variables, although in actuality there is no difference in models except in the way in which engine airflow is treated to identify compressor pumping capacity changes. Theoretically, there are only minor differences between the two models in terms of results, and these differences have been found to be relatively inconsequential. On the practical side, several differences have been noted.

An error analysis for the prototype AIDAPS gas path analysis was completed for both the N1C-independent and the P3C-independent approaches. Based on an earlier analysis, instrumentation repeatability requirements were specified on the basis of the degree of variation in diagnostic model results that can be tolerated without affecting the ability to reliably and accurately detect and isolate engine component degradation (roughly, variations no larger than about ±1 percent). The analysis results show that the prototype system conforms to that requirement.

In the case of the independent variables, the error effects are almost identical for the two cases. With the employment of an IGV correction procedure and improved baseline calculation technique, data collection results for the UH-1/T-53 also show no significant difference in scatter for the two approaches. In view of this and the advantages afforded by the greater reliability of the N1 measurement, N1 appeared as the clear choice for independent variable, and that choice was made.

### 3.2.1.3 Development of Confidence Level Estimates for Diagnostic Results and Sensor Diagnostics

One of the attractive features of the AIDAPS gas path monitoring approach is that the quality of the measurements are to a large extent reflected in the diagnostic results. For example, no single sensor error can cause a diagnostic

result that appears valid. This is true because the physically possible individual and combined variations of the engine component efficiencies and other variables are limited. In general, a sensor error produces an imbalanced result; for example, several sensor errors can cause an apparent drop in GP turbine efficiency, but at the same time will cause an apparent increase in compressor efficiency. While small increases in compressor efficiency are possible when turbine efficiency degrades (due to rematch), increases comparable in magnitude to the decrease in turbine efficiency indicate a sensor error.

Sensor diagnostics takes place on two levels. Gross sensor errors or malfunctions are identified by means of credibility limits applied directly to the measured parameter variations from baseline. Logic for detecting and identifying smaller, but significant, sensor problems is based on the use of the confidence level check for detection; single-fault pattern recognition can be implemented for isolation of sensor errors.

### 3.2.2 Vibration Logic

#### 3.2.2.1 Logic Determination Procedure

The vibration fault detection logic was developed for the UH-1H engine and gear boxes using the logic determination logic described in the following paragraphs:

Logic determination starts with verified vibration implant data and corresponding verified baseline data. The verification is done by using 1024-point PSD plots; it determines the validity of the data and provides a gross indication of the detection of the implant.

Following data verification, the corresponding 128-point implant and baseline, PSD data are then analyzed by the likelihood analysis program. This determines the relative indicativeness of the various bands and provides useful statistical data.

Based on the detection indicativeness of the individual bands and other factors in the frequency band selection criteria, a set of frequency bands are selected. The information in the selected frequency bands is then combined to form a discrimination indicator. Two discrimination indicators are used in the present logic--one for bearings and the other for gears. The discrimination indicator is a weighted sum where the weights are based on the standard deviation of the baseline data. The composite indicator is the sum of the absolute changes. This, then, is a measure of the magnitude of the multi-dimensional pattern vector of the selected frequency bands.

The preliminary formulation of the discrimination indicator is evaluated by using actual test data. This also is a verification of the derived logic. Such items as start/stop conditions, instrumentation requirements, display message level and the intended use of the output are evaluated. The finished form of the logic is documented in conventional flow diagram form.

### 3.2.2.2 Fault Discrimination

Basically, the vibrational faults are detected by comparing the value of a discrimination indicator to a predetermined threshold value. Due to the significant difference between gear and bearing fault signatures, two separate discrimination indicators are used.

The gear fault discrimination indicator is selected as the norm of the vector formed from the normalized composite modulation index changes at the gearmesh frequency and the first harmonic of the gearmesh frequency. The modulation phenomenon of a gear fault was isolated by the likelihood analysis of the gear implant data and was confirmed through the modulation index analysis of the original narrowband PSD data. The gear fault discrimination indicator is mathematically defined as

$$(D.I.)_G = \frac{1}{2} \sum_{i=i}^{2} \left| \frac{M_{CD} - \overline{M_{CB}}}{\sigma_{MCB}} \right|_i$$

where

$M_{CD}$ = composite modulation index of the test data

$\overline{M_{CB}}$ = average value of the composite modulation index of the baseline data

$\sigma_{MCB}$ = standard deviation of the baseline composite modulation index

$M_{C.}$ = composite modulation index

$$= \sum_{j=-5}^{+5} \frac{\left(A_{SBj}\right)}{A_{CR}}$$

$A_{SB}$ = amplitude of the sideband frequency

$A_{CR}$ = amplitude of the carrier frequency

$i = 1$ $M_{C.}$ at gearmesh fundamental frequency

$i = 2$ $M_{C.}$ at 1st harmonic of gearmesh frequency

$j$ = jth sideband about the carrier frequency

The composite modulation index, $M_{C.}$, represents the overall ±5 sideband amplitude distribution versus the carrier amplitude. The increase in composite modulation index represents the transfer of vibrational energy from the carrier frequency to the sideband frequencies as a consequence of the gear fault.

Because of the dependence of these frequencies on rpm, rpm-tracking is required for the gear discrimination indicators.

The bearing fault discrimination indicator is the norm of an $N_B$-dimensional vector of normalized amplitude changes of a selected set of frequency bands. The frequency bands are primarily determined by likelihood analysis of the test data. Mathematically, the bearing fault discrimination indicator is defined as:

$$(D.I.)_B = \frac{1}{N_B} \sum_{i=i}^{N_B} \frac{|A_D - \bar{A}_B|}{\sigma_B}$$

where

$A_D$ = amplitude of the test data

$\bar{A}_B$ = average amplitude of the baseline data

$\sigma_B$ = standard deviation of the baseline amplitudes

$N_B$ = number of frequency bands involved

$i$ = ith frequency band

The discrimination indicator represents the magnitude of the normalized amplitude changes of the frequency bands which are sensitive to bearing faults.

The use of normalized change over baseline with respect to the standard deviation of the baseline data provides a direct measure of the significance of the change at that particular frequency band.

In addition to the gear and bearing discrimination indicators, the engine rotor unbalance is monitored by two separate rotor unbalance discriminators indicators, i.e., $(DI)_{u1}$ and $(DI)_{u2}$, for monitoring the unbalance of $N_1$ and $N_2$ respectively. Because of the narrow bandwidth of the unbalance spectrum, binary discrimination indicators are used to improve the sensitivity of detection. The rotor unbalance discrimination indicators are defined as follows:

$$(DI)_{ui} = (DI)_{ui1} \bigvee (DI)_{ui2}$$

where $(DI)_{ui}$ is the unbalance discrimination indicator for $N_i$, $i$ = 1 or 2

$(DI)_{ui1}$ is the component unbalance discrimination indicator at the fundamental rotor unbalance frequency.

$(DI)_{ui2}$ is the component unbalance discrimination indicator at the first harmonic of the rotor unbalance frequency.

$(DI)_{uij}$  j = 1 or 2 will take a binary value of "1" or "0" according to the following equation:

$$(DI)_{uij} = \text{"1" if } \frac{A_{Di}(j) - \overline{A_{Bi}(j)}}{\sigma_{Bi}(j)} \geq T_u$$

$$= \text{"0" otherwise}$$

$\bigvee$ is the logical "or" operator.

$A_{Di}(j)$ is the current measured vibration amplitude from band j.

$\overline{A_{Bi}(j)}$ is the average baseline vibration amplitude from band j.

$\sigma_{B_i}(j)$ is the standard deviation of the baseline vibration amplitudes from band j.

$T_u$ is the unbalance detection threshold.

Because the rpm dependence of the rotor unbalance frequency and its first harmonic, rpm-tracking is required for the rotor unbalance monitoring discrimination indicators.

3.2.2.3 <u>Determination of Detection Threshold</u>

The detection threshold is a number to be compared with the value of the discrimination indicators. Since the discrimination indicators are represented by the changes in terms of number of $\sigma$'s of the baseline data, the detection threshold is related to the false alarm probability of the detection of faults. The lower the detection threshold is set, the higher the false alarm probability. This is depicted in Figure 3-6. For normally distributed data, the detection threshold equivalent to the $3\sigma$ level of the individual bands or composite modulation indices provides a confidence level of 99 percent for the detection.

3.2.2.4 <u>Reduction of False Alarms by Majority Logic</u>

False alarms are caused by the occurrence of signal amplitudes from good components which exceed the fault detection threshold. The occurrence of these signals is statistically infrequent as opposed to the continuous occurrence of signal levels due to a faulty component; therefore, the system false alarm rate can be reduced significantly by the use of majority logic without affecting the fault detection capability of the system. The majority logic chosen for AIDAPS requires a majority of three threshold exceedances out of four consecutive tests. Using this approach, the false alarm probability with majority logic $P_M(F)$ is

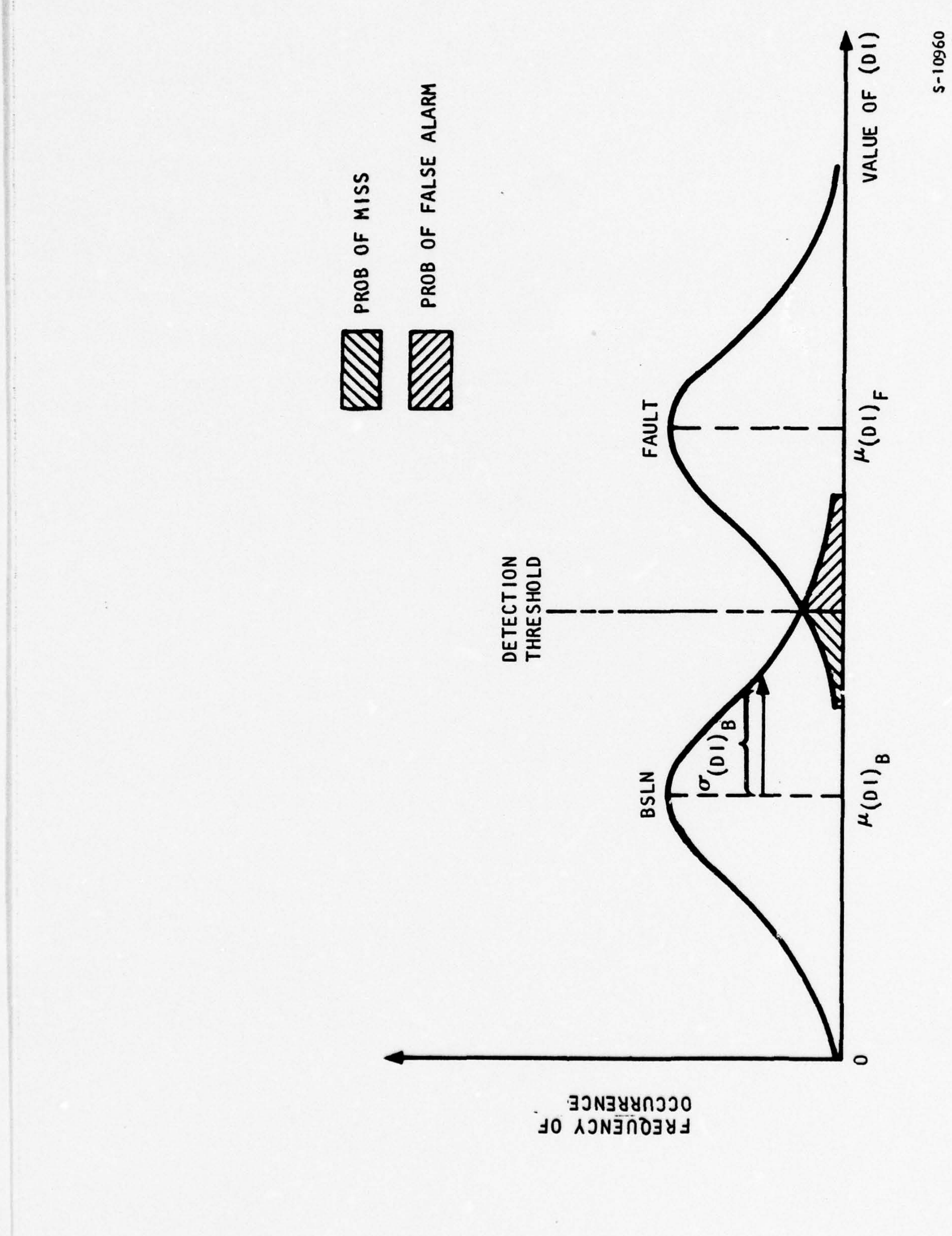



Figure 3-6. Probabilistical Determination of Detection Threshold

$$P_M(F) = P(F)\ [3P(F)^2 - 2P(F)^3]$$

where $P(F)$ = False alarm probability without majority logic

Assuming the threshold is set to where the false alarm rate for a single test, $P(F)$, is one percent, this majority logic approach produces false alarm probability for each component of:

$$P_M(F) = 0.01\left[3(0.01)^2 - 2\ (0.01)^3\right]$$
$$= 0.000003$$

For the components monitored on the UH-IH aircraft by the prototype system, the expected vibration false alarm rate for the aircraft is:

Vibration system

false alarm rate ≅ 1 false alarm per 2000 hours of flight.

3.2.2.5 Frequency Band Selection

The selection of frequency bands for gear fault monitoring is straightforward. It follows directly from the likelihood analysis and modulation index analysis results, i.e., ±5 sidebands around the gearmesh fundamental frequency and its first harmonic.

The selection of the frequency bands for bearing fault monitoring is more involved because of the many bearings involved and the many failure modes associated with each bearing. The following general criteria are used for the selection of the monitored frequency bands:

(a) Fault discrimination capability, as represented by the likelihood index value of the frequency band

(b) Repeatability of discrimination capability at various power levels and test conditions

(c) Consistency of signature bands among implants of the same type

(d) Vulnerability to gearmesh interferences

(e) Maximum commonality of frequency bands among many implants to minimize the total number of frequency bands required

(f) Choice of wide bands as compared with discrete single frequencies in order to reduce false alarm rate, reduce the probability of a miss, and to allow wider bandwidths for easy hardware implementation

(g) Selection of more bands for higher failure rate or more critical failure implants, and fewer bands for lower failure rate or less critical implants

Using these criteria, gear and bearing logic was developed for the UH-IH engine, transmission, hanger bearing, 42-deg gear box, and 90-deg gear box.

### 3.2.2.6 Vibration Monitoring Start/Stop Conditions

The vibration monitoring start and stop points are initiated by values of $N_1$ and $N_2$ rpm's. Since the fault detection is for two levels, i.e., maintenance and flight safety, a different set of start and stop points is associated with each level.

The start/stop condition for maintenance level detection is as follows:

Start - if $N_1 \geq 85\%$ and $N_2 \geq 94\%$

Stop - otherwise

If engine rpm is below these limits, maintenance fault monitoring will not be performed. Actually, this condition seldom occurs in flight (less than 10 percent of the flight time). Also, the transitory nature and instability of rpm in this range would necessarily complicate the logic.

The start/stop condition for flight safety detection is as follows:

Start - if $N_1 \geq 71\%$ and $N_2 \geq 64\%$

Stop - otherwise

This rpm range includes ground idle.

### 3.2.2.7 Vibration Logic Summary

The final vibration logic was developed as a result of the analysis of data collected during Phase I. The vibration sensor locations and faulty part implants were determined by a cooperative effort of AiResearch, Lycoming and Bell Helicopter. The likelihood analysis of collected data performed at AiResearch allowed a selection of frequency bands to be used for fault detection logic for each sensor. Discrimination index limits for each sensor were determined empirically during the Phase V testing of implants at Ft. Rucker. The details of vibration logic are summarized in Appendix E. A page is provided for each of the thirteen sensors in a prototype system. The information presented for each sensor consists of (1) the sensor number, type, and location, (2) the component parts monitored, (3) the selected frequency bands and band widths for bearing and gear fault detection and rotor out-of-balance detection, and (4) the discrimination index limits for diagnostic and flight safety logic.

### 3.2.3 Functional/Mechanical Logic

Diagnostic logic for functional/mechanical systems was derived from the recommended logic provided by Bell Helicopter Company and by AVCO-Lycoming Division. The diagnostic logic was submitted in July 1974. This

logic was reviewed and revised as required based on the results of the analysis of data collected in the engine test cell and other information sources described in Section 3.1.4 of this report. Extensive implant testing was not considered necessary to the development of the functional/mechanical logic due to the straightforward and simple nature of the logic and its implementation. Nevertheless, during initial testing of the prototype system hardware it was discovered that certain logic functions were not performing properly due to unanticipated aircraft startup procedures and sensor environmental conditions. This resulted in additional modification to the logic. The resulting functional/mechanical logic provided the fault detection capability defined in Table 3-3.

The aircraft parameters which must be monitored to implement the functional/mechanical logic are listed in Table 3-4.

The parameters are grouped according to aircraft subsystem. Several of the engine parameters are used for gas-path analysis and are identified. The last column identifies those sensors which must be added to the existing aircraft instrumentation but does not apply for those which are required for gas-path analysis. Thus, this column identifies the net added sensors for functional/mechanical monitoring.

## TABLE 3-3

## FUNCTIONAL/MECHANICAL FAULT DETECTION CAPABILITY

| Component | Part | Detected Fault | Comment |
|---|---|---|---|
| Engine | IGV/fuel control | Rigging/IGV command | |
| Engine | Bleedband/fuel control | Rigging/bleedband command | |
| Engine | Oil system | Oil level low or lines clogged on relief valve or pump | |
| Oil cooler blower | | Failure to circulate sufficient air to cool oil | |
| Engine | Fuel nozzles | Plugged or eroded sufficiently to cause bad EGT pattern | |
| Transmission | Oil temperature transmitter | Failure "high" | Failure "low" may be detected in some cases. |
| | or oil temperature switch | Failure "on" | Failure "off" detectable if temperature goes high. |
| Transmission | Oil cooling components | Clogged lines or thermal bypass valve or oil pump | |
| Transmission | Oil pressure transmitter or oil pressure switch | Failure low or failure high<br>Failure on or off | |
| Transmission | Relief valve or oil pump | Relief valve adjusted improperly (low) or oil pump failure | |
| Transmission | Relief valve | Adjusted too high or failed closed | |
| Transmission | Internal Oil filter | Clogged or switch failure | |
| RPM warning Control Box | RPM warning Control Box | Fails to generate audio command signal for low RPM.<br>Fails to generate light command signal for low RPM.<br>Fails to generate light command for high RPM. | |

TABLE 3-4

FUNCTIONAL/MECHANICAL PARAMETERS

| Aircraft Subsystem | Parameter | Analog or Discrete | Used for Gas Path | Added Sensor |
|---|---|---|---|---|
| 03 | CIT | A | X | NA |
| | EGT average | A | X | NA |
| | EGT pattern (6 measurements) | A | | |
| | N1 | A | X | NA |
| | N2 | A | X | NA |
| | NR | A | | |
| | Torque pressure | A | X | NA |
| | IGV | A | X | NA |
| | Bleedband | D | X | NA |
| | Bearing No. 2 oil scav. temp. | A | | X |
| | Engine oil pressure | A | | |
| | Engine oil temperature | A | | |
| 04 | Transmission oil temp. | A | | |
| | Transmission temp. switch | D | | |
| | Transmission oil pressure | A | | |
| | Transmission oil pressure switch | D | | |
| | Internal oil filter $\Delta P$ | D | | X |
| 09 | RPM warning control box power | D | | |
| | Control box audio indicator output | D | | |
| | Control box light indicator output | D | | |
| Chip Detector Signals to Supplement Vibration Fault Detection | | | | |
| 03 | Engine accessory GB chip detector | D | | X |
| | Engine brg No. 2 chip detector | D | | X |
| | Bearing No. 3/4 chip detector | D | | X |
| 04 | Transmission oil sump chip detector | D | | |
| | $42^{\circ}$ gearbox chip detector | D | | |
| | $90^{\circ}$ gearbox chip detector | D | | |

# 4. PHASE III - DESIGN

## 4.1 TASK 1: HARDWARE DESIGN

Detailed design information has been documented throughout the program and submitted separately, and as part of the Interim Technical Reports in accordance with the requirements of the CDRL. In particular, outline drawings, photographs of hardware, parameter lists, parameter transfer equations, and descriptions or preliminary specifications of the various system end items are provided in References 3, 4, and 5. Detail drawings, including schematic diagrams, were submitted under CDRL Data Item A008. An error source study, identifying the sources and magnitude of possible system errors, was submitted as a part of the Phase III statement-of-work requirements.

### 4.1.1 System Engineering

The system engineering of AIDAPS was based on information derived from a number of sources. These include those discussed in Section 3 of this report; namely, BHC, ALD, and data collected by AiResearch at Ft. Rucker, Alabama. Additional information relating to system design aspects, such as candidate systems and parameters for monitoring, ranking of parts according to failure rates and criticality to flight; and desirable system features such as system partitioning, flexibility, and maintenance message format and display was sought from the AVSOM Concept Development Branch, AVSCOM System Reliability Group, TRADOC, and several maintenance groups at Ft. Rucker, Alabama, Ft. Hood, Texas, and Ft. Eustis, Virginia. Based on the available information, a system design was developed which detects and annunciates degraded performance of the major components of the UH-1H helicopter.

#### 4.1.1.1 Fault-Detection Capability

The fault detection capability of AIDAPS for the UH-1H is described in the following table.

| Components | Parts | Detected Fault |
|---|---|---|
| Engine | Compressor | Erosion, Fouling, FOD |
| | Diffuser | Erosion, Fouling, FOD |
| | Turbine | Erosion, Fouling, FOD |
| | Fuel Nozzles | Coked, Eroded |
| | Main Bearing | Outer Race and Ball Spalls |

(Continued)

| Components | Parts | Detected Fault |
|---|---|---|
| Engine | VIGV and Bleed Band | Mis-scheduling, Broken Linkages, Fuel Control Errors |
| | Oil Pump | Low Pressure Detected |
| | Oil Cooler | High Temperature Detected |
| Transmission | Input Gears | Scoring |
| | Sun Gears | Scoring |
| | Nine Most Critical Bearings | Ball, Roller, or Race Spalls |
| | Oil Pump | Low Pressure Detected |
| | Oil Cooler | High Temperature Detected |
| $42^0$ Gearbox | All Bearings<br>Both Gears | Ball, Roller, or Race Spalls<br>Scoring |
| $90^0$ Gearbox | All bearings<br>Both Gears | Ball, Roller, or Race Spalls<br>Scoring |
| Shaft Hanger | Bearing | Ball or Race Spalls |

Other Aircraft Condition Monitoring

- Engine/Rotor Overspeed
- RPM Warning Control Malfunctions
- Engine/Transmission Over-Torque
- Engine Over-Temperature

#### 4.1.1.2 Description of System and Operation

The UH-1H AIDAPS consists of an airborne segment and a ground segment as depicted in the block diagram shown in Figure 4-1. The airborne segment consists of (1) transducers, (2) a data acquisition unit (DAU), (3) a computer memory unit (CMU) or digital data recorder (DDR), and (4) a cockpit annunciator panel. The ground segment consists of (1) a diagnostic analyzer (DA), and (2) a digital printer. The system can be used in either of two configurations: Hybrid I, which includes the CMU and incorporates onboard fault detection and diagnostics; and Hybrid II, which substitutes the digital recorder for the CMU and provides airborne data acquisition with ground processing. The system is modularly interchangeable between the Hybrid I and

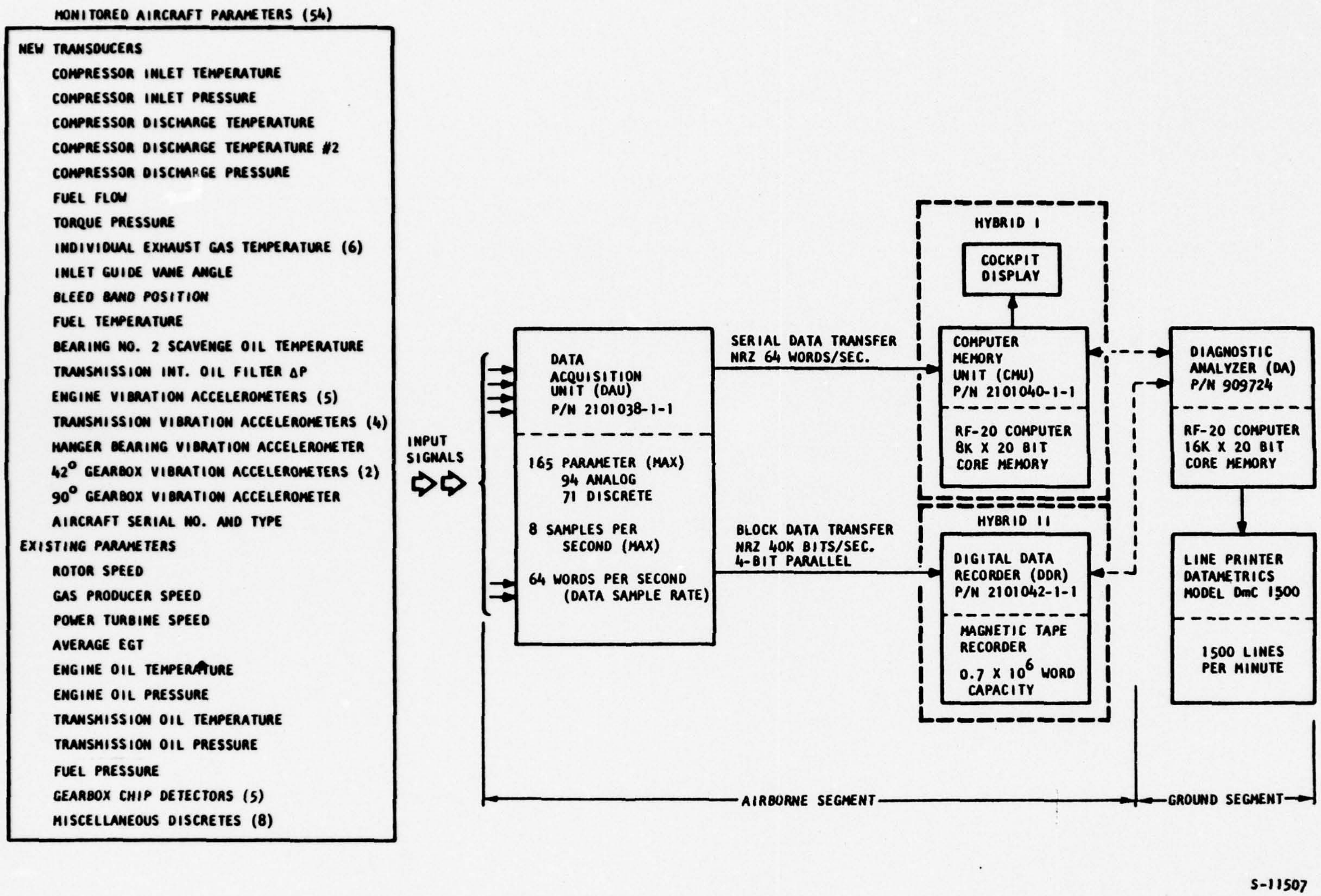



Figure 4-1. AIDAPS System Block Diagram

Hybrid II configurations by means of directly interchanging the digital recorder and the CMU; they are mechanically interchangeable without modification to the aircraft or mounting hardware. The fundamental difference between the two configurations is in information availability--Hybrid I performs data acquisition, preprocessing, and a portion of the diagnostics in flight, whereas Hybrid II relies upon the ground segment for all diagnostic computations. For both configurations data printout is performed in the ground segment.

For the Hybrid I configuration, information is output from the system at three levels:

(a) To the pilot by means of the cockpit annunciator panel, which gives an in-flight warning of either a detected fault which requires maintenance, or a critical fault which could impact flight safety. Additionally, the annunciators indicate whether the fault is in the engine or power train. A fourth annunciator indicates the necessity to transfer data from the airborne segment to the ground segment. Data transfer is required during baseline operations, when the DDR is filled to capacity, and when a fault detection event occurs.

(b) To the pilot or maintenance crew by means of a 3-digit display on the front panel of the CMU. This display, which is accessible to the crew when the aircraft is on the ground, will allow them to determine more specific information about faults which are detected in flight by the CMU; it is intended to be used immediately after landing, when a fault has been indicated by the cockpit annunciators, and provides a level of self sufficiency to the aircraft (i.e., some of the diagnostic results will be available without processing at the DA). Appendix B tabulates the message codes used for the CMU display and the corresponding messages which are printed out when data is processed at the DA. If more than one fault is detected by the CMU, they may be read out sequentially by pressing the "advance" button. Additionally, three latching annunciator flags are located on the CMU front panel to indicate failures of AIDAPS components (sensors, DAU, or CMU) which are detected by the self-test and diagnostic computer program subroutines contained in the CMU.

(c) To the ground crew by means of the DA and printer. The DA performs a portion of the diagnostic processing for Hybrid I. The results of processing is output in the form of a hard copy printout. The CMU message codes and the DA messages are indicated in Appendix B. This appendix also lists additional messages which are output via the printer as a result of DA processing. Note that additional messages will be printed in the event of a DA operator control sequence error.

For the Hybrid II configuration, the transfer data light is activated in the cockpit when the recorder is filled to capacity. This calls for processing of the tape-recorded data at the DA. No other outputs are available on

the aircraft. All other information is output to the ground crew by means of the DA and printer. The DA performs all diagnostics and provides all of the functions which are accomplished by the CMU and the DA in a Hybrid I configuration except that they are performed on the ground. The printout format and information content is identical for both configurations.

An important feature of the system is the aircraft independence from any ground station. The characterization data for each aircraft is stored onboard--in the CMU core for Hybrid I, and on the digital data tape for Hybrid II. This characterization data consists of the serial numbers and elapsed operating times for principal LRU's, the baseline data which characterizes the normal operating performance for those LRU's, and any trend history data accumulated for those LRU's. This allows any aircraft to operate in conjunction with any ground station. Each time that data from a CMU or a DDR are processed at a DA, the updated data is reloaded into the CMU or DDR.

In addition to processing flight data, the DA provides the capability to compute baseline functions for LRU's from flight data. It also provides a facility for entering LRU serial numbers, time-in-service, etc., when LRU's are initially installed or changed.

In normal operation of the DA, two control modes are used. After a flight when DA processing is desired, the tape recorder or CMU is connected to the DA. The control mode thumbwheels are set to "00" (Normal Data Processing). The date thumbwheels are set and the tail number of the aircraft from which the airborne unit was removed is entered via the appropriate thumbwheels. When the "Execute Processing" button is pushed, the DA reads the data, processes it, and prints fault or error messages, as required. When processing is complete "End of Processing" is printed. If neither faults nor errors are indicated, the control mode thumbwheels are set to "58" (Normal Rewrite Mode). The updated header data including updated baselines is automatically transferred to the CMU or DDR. After "End of Processing" the CMU or DDR is disconnected and reinstalled on the aircraft.

If fault messages are printed, the appropriate maintenance action should be taken. When new (or overhauled) components are installed on the aircraft, the component serial numbers and "times in service" (logistic data) must be loaded as header data. This is accomplished by switching the aircraft selector switch to "Individual" with the correct date and aircraft identification numbers selected. Each LRU number, serial number, and operating time is inserted sequentially by setting the "Logistics Data Input" thumbwheels and pressing "Insert" for each set. If only one component is changed, only that LRU logistics data should be entered. The LRU identification codes are as follows:

| | |
|---|---|
| 00 | Aircraft Ident.--used only if a new aircraft is substituted. |
| 01 | Engine |
| 02 | Transmission |

03 Hanger Bearing

04 $90^{0}$ Gearbox

05 $42^{0}$ Gearbox

After entering this data and writing it as a header onto the peripheral, the peripheral is installed in the aircraft for baseline formulation.

### 4.1.2 Airborne and Ground Segment

The design of the airborne and ground segment hardware was described in detail in Reference 5. Figure 4-2 shows the DAU, CMU, and cockpit display portions of the Hybrid I airborne segment. A change to the Hybrid II configuration is achieved by replacing the CMU with the DDR. The remainder of the hardware consisting of sensors and wiring are distributed throughout the aircraft and are described in Reference 5 and CDRL Data Item A008.

Figure 4-3 shows the ground segment as installed in the trailer facility at Ft. Rucker, Alabama. The ground segment consists of the DA and the Datametrics model DmC1500 line printer. Figure 4-3 shows the CMU connected to the DA as it would be for normal processing of data.

### 4.1.3 Prototype System Software

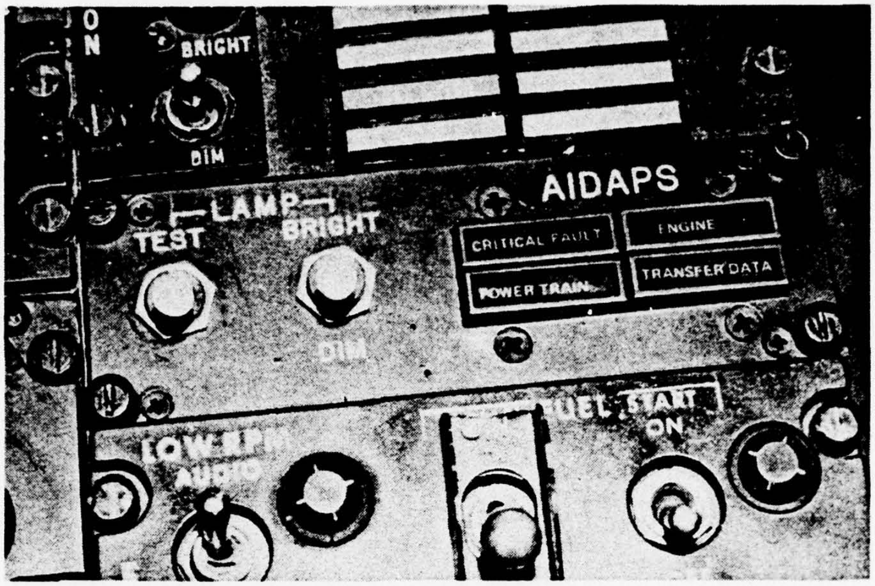
The prototype system software consists of computer programs for the CMU and the DA. These programs handle all gas path, vibration, and mechanical/functional processing and all housekeeping functions necessary to accomplish the fault detection and annunciation functions of AIDAPS. This software was described in Reference 6 and program documentation was submitted under CDRL Data Item A013.

The gas path and vibration processing are the more important and unique parts of the system software and are briefly described below. The mechanical/functional and housekeeping parts of the program use straightforward and conventional programming techniques and are not described herein.

#### 4.1.3.1 Gas Path Data Processing

Gas path data processing is depicted in Figure 4-4. The processing functions are shared between the CMU and the DA in Hybrid I and performed entirely in the DA in Hybrid II.

Gas path sensor data must first satisfy certain stability and credibility criteria before being used for engine condition monitoring purposes. This assures that dynamic engine conditions or faulty sensor signals will not cause false indications of degraded engine performance. Whenever a new engine is installed for use, baseline performance data are generated by evaluating gas path sensor data in the thermodynamic model during the first few hours of operation. This is an automatic procedure which ends when the baseline is mature and is stored in the computer memory. Continuing evaluation of gas path sensor data produces post baseline results representing current engine




COCKPIT DISPLAY

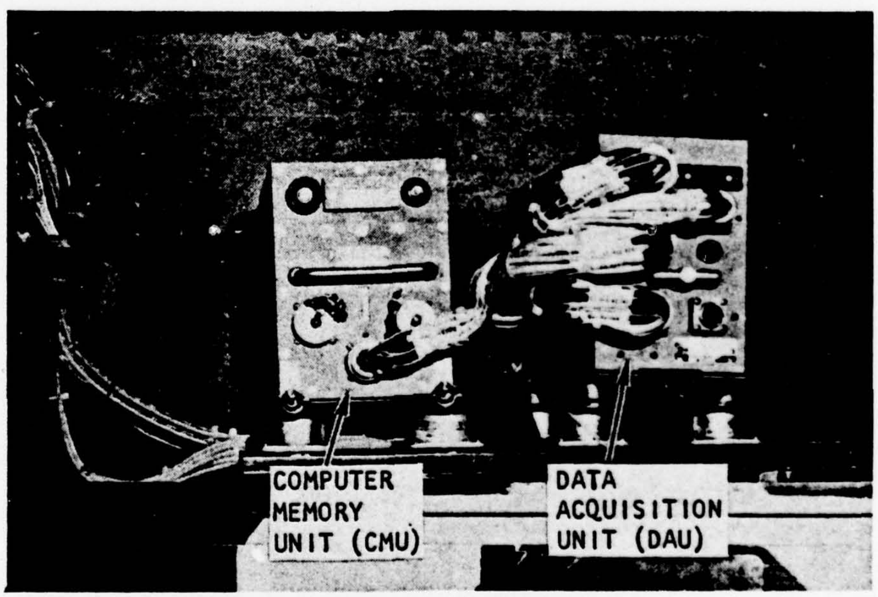



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Figure 4-2. Airborne Segment

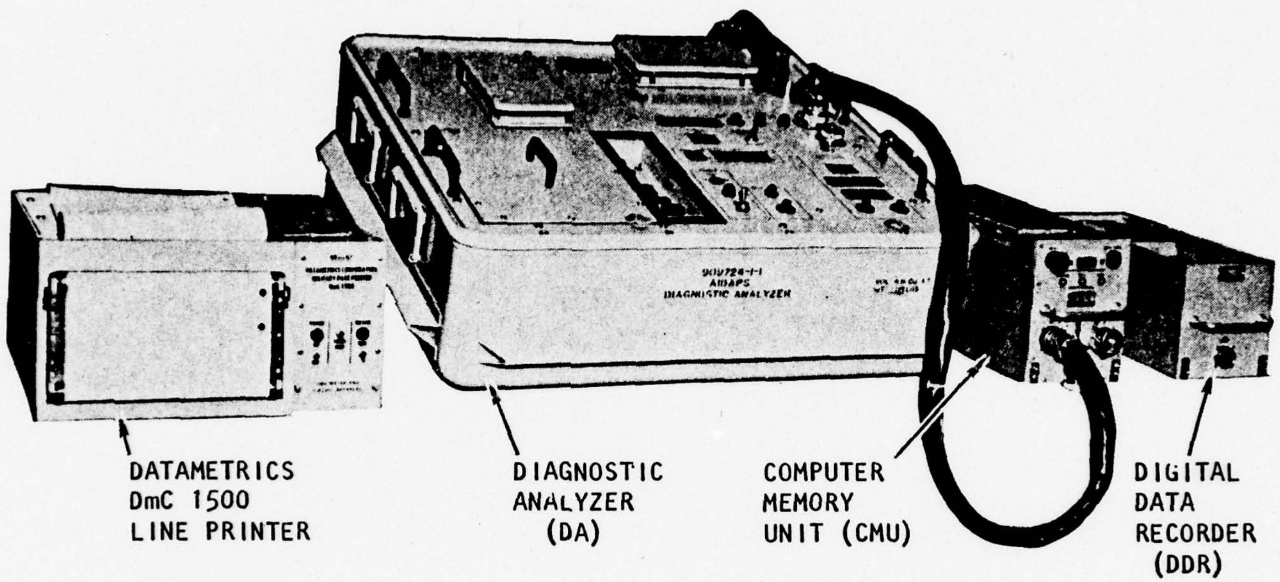


Figure 4-3. Ground Segment

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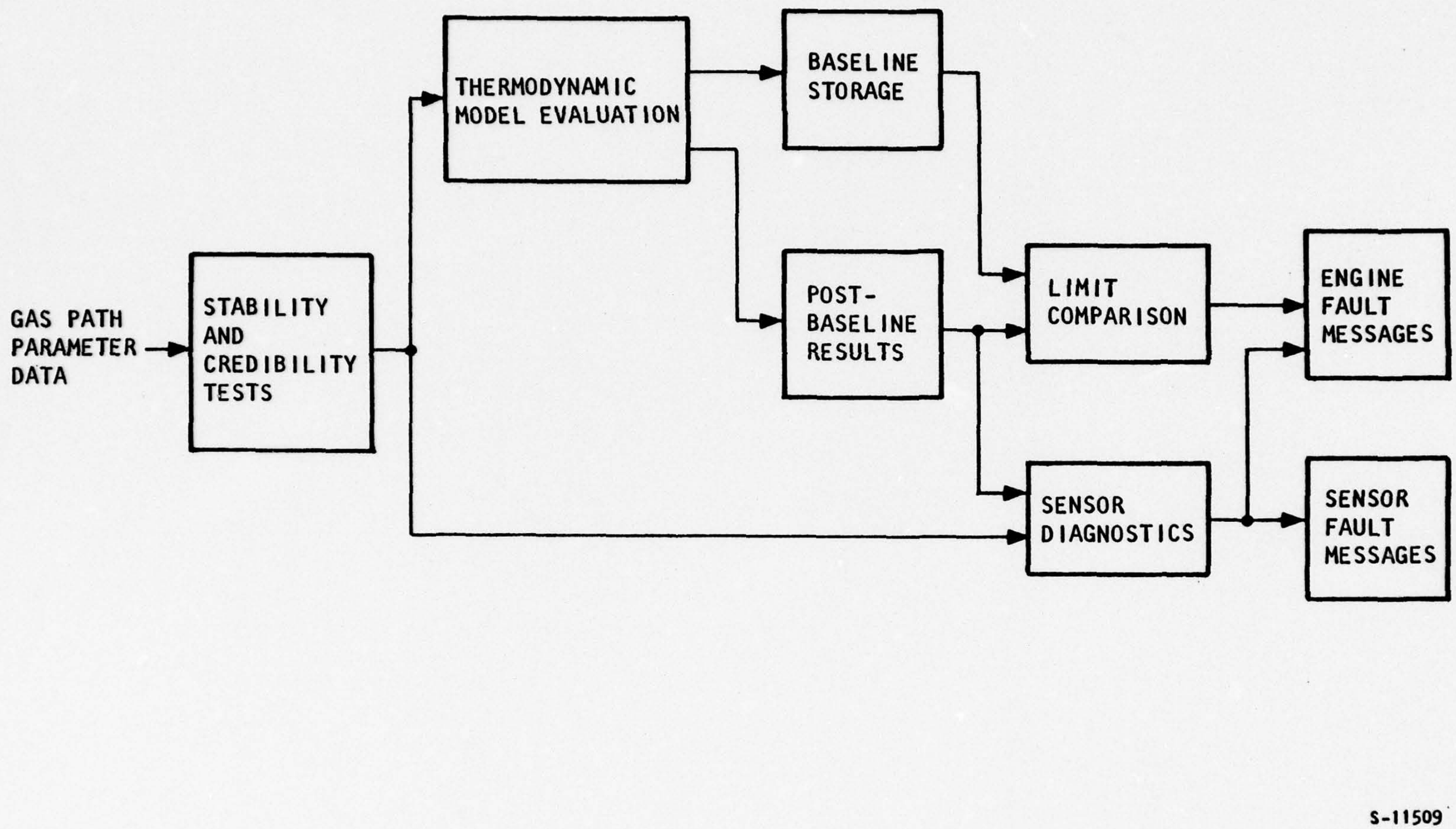



Figure 4-4. Gas Path Software Flow Diagram

performance. The baseline and post baseline performance data are compared and when the differences between certain performance characteristics exceed pre-established limits, a faulty engine condition exists. An analysis of which performance characteristics have changed allows isolation of the fault to a particular engine module. Simultaneously, sensor diagnostic tests are performed to indicate whether the degraded condition is possibly due to sensor error. Thus, when a degraded condition is detected, an appropriate engine fault message can be prepared for display. If the degraded condition indication is caused by a sensor error, the engine fault message is inhibited and a sensor fault message is prepared instead. Finally, messages are printed for maintenance actions to be taken by aircraft maintenance personnel.

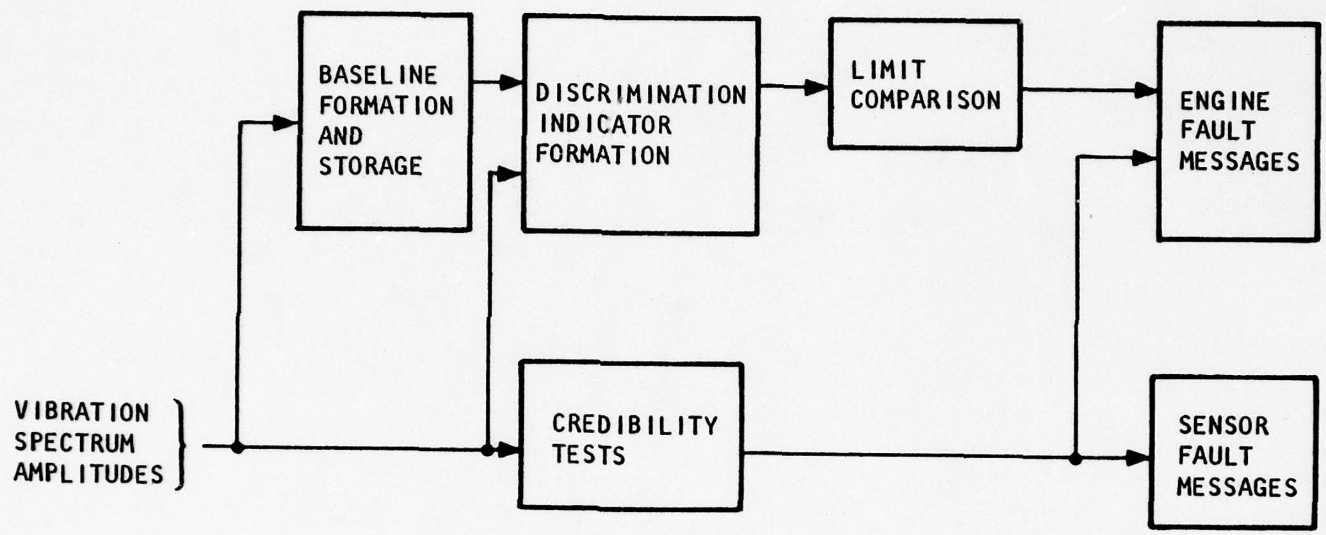
### 4.1.3.2 Vibration Signal Processing

Vibration signal processing is depicted in Figure 4-5. Spectral analysis of accelerometer signals is performed in the DAU. Subsequent processing of vibration spectrum signal amplitudes is performed in a similar manner in either the CMU or the DA, depending on whether the Hybrid I or Hybrid II mode is used. Baseline vibration amplitudes are formed and stored during the first few hours after a new engine transmission, or gearbox installation. Vibration signal amplitude information is used to form discrimination indicators which indicate change from baseline condition. A discrimination indicator which exceeds a pre-established limit indicates component part degradation which requires maintenance action. Analysis of the various discrimination indicators allows isolation of the faulty condition and produces appropriate fault messages. Signal credibility tests detect sensor failures and produce sensor fault messages and inhibit any false indications of degraded aircraft component condition.

The condition monitoring techniques described earlier were developed to be fully automatic, including the generation of baselines and changeover to the fault detection mode of operation. The software recognizes engine startup and shutdown events and prevents erroneous messages during these periods. Vibration fault messages requiring maintenance personnel action were intentionally kept short and simple. However, the software provides a system interrogation capability which produces additional intermediate data and results that can be useful in trending and trouble shooting.

## 4.2 TASK 2: AIRCRAFT INSTALLATION

Task 2 covered the installation of sensors and wiring modification kits on four UH-1H helicopters. Details of the installation are described in Reference 5. Photographs of installed airborne and ground segment hardware are shown in Figures 4-2 and 4-3.




S-11508

Figure 4-5. Vibration Software Flow Diagram

# 5. PHASE IV - PROTOTYPE APPLICATION

## 5.1 TASK 1: PROTOTYPE FABRICATION

Prototype hardware fabricated during the contract period was delivered either to Ft. Rucker, Alabama, or transferred to the government bond room at AiResearch in accordance with contractual requirements.

The following prototype hardware was delivered to Ft. Rucker, Alabama.

6 Aircraft modification kits

4 Data acquisition units (DAU)

2 Computer memory units (CMU)

2 Digital data recorders (DDR)

2 Diagnostic analyzers (DA)

1 DAU test set

The following prototype hardware was transferred to the government bond room in a partially assembled/checked-out condition:

2 DAU's

2 CMU's

1 DAU test set

All production test equipment has been retained by AiResearch to support maintenance and repairs when necessary.

## 5.2 TASK 2: PROTOTYPE INSTALLATION

The installation and checkout of both Hybrid I and Hybrid II configurations of prototype airborne segments were accomplished on all four UH-1H aircraft during January 1976.

The installation and checkout of the ground segment in both configurations was also accomplished during January. During subsequent performance testing several software deficiencies were discovered and corrected as required.

Details of prototype hardware installations, including a description of checkout procedures and test results, are presented in Reference 8.

## 6. PHASE V - TEST AND SUPPORT

This section summarizes the results of all of the prototype system tests, including bench tests, at AiResearch in Torrance, California, and airborne testing at Cairns AAF, Ft. Rucker, Alabama. The prime objective of this phase of the program was to assess the detection capability of the Hybrid I and Hybrid II prototype AIDAPS configurations. Although minor differences were noted on some individual tests, test results showed the detection capability of the two configurations of AIDAPS to be essentially the same. Consequently, for evaluation purposes, the noteworthy differences are those which will affect how the systems are used in operation: e.g., Hybrid I with on-board processing and display provides real-time assessment of helicopter condition and immediate maintenance information after landing; Hybrid II provides the same information, but only after processing the recorded data at the diagnostic analyzer.

Two major parts of airborne testing were the performance test, which had a duration of four months, and the demonstration test, which had a duration of six months. In the performance test, implants were selected and tested to determine how well the system detected these parts and to allow system/logic modifications to improve system performance. During this test period 29 degraded parts were tested, and after the recommended system/logic modifications were accomplished, a degraded part detection score of 89 percent was achieved.

During the demonstration test, degradation conditions were simulated which were unknown to AiResearch. It was desired that a large number of degraded part tests be performed to provide a large data base for assessing the detection capability. About two hundred such degraded part tests were simulated. This required a large number of component maintenance actions on the four aircraft used for the test. During six months of testing, 520 component replacements were accomplished. Although this large number of maintenance actions invalidated some of the results, the objective of determining the detection capability of AIDAPS was accomplished. The overall score for detecting the presence of degraded parts during the demonstration test was 74 percent.

These airborne testing results served to demonstrate conclusively the capability of AIDAPS for detecting a wide variety of degraded part conditions in all of the UH-1H power train components. The tests also allowed the reliability of the prototype hardware to be assessed. After correction for elimination of failures due to correctable design deficiencies, an MTBF of 945 hr is projected for the Hybrid I airborne segment of the system.

The following paragraphs provide added details of these tests.

## 6.1 TASK 1: BENCH TEST

Static system and environmental bench tests were performed at AiResearch, Torrance prior to shipment to USAADTA, Ft. Rucker, for airborne testing. The static system tests checked the performance of the individual units (DAU, CMU, and DA), the interfaces between these units, and the logic implemented in the CMU and the DA. These tests included conducting ATP's and recording and processing simulated flight data. Test results indicated that system performance was adequate to begin airborne performance testing. The second type of bench tests was environmental tests of two airborne units--the DAU and the CMU. The DDR was not tested since it is already qualified to the environmental performance requirements of MIL-E-5400. The DAU tests indicated correct operation between -54°C and +60°C (+71°C required). The CMU tests indicated correct operation between -18°C and +71°C (-54°C required). Because of the urgency to begin Phase V airborne tests, design modifications to achieve correct operation over the full temperature range were not accomplished.

## 6.2 TASK 2: AIRBORNE TESTING

Flight safety qualification tests were completed in January 1976, after which the performance and demonstration tests were conducted using implanted, degraded parts to simulate naturally degraded components. The test results demonstrate the capability for detecting degraded part condition and also indicate the feasibility of fault isolation to component modules. The parts tested included naturally degraded parts obtained from Army overhaul depots where they were removed from components returned for repair, and artificially degraded parts created especially for the AIDAPS program. Obviously the ability of the system to detect defective parts is dependent on the level of degradation of the part being tested. For example, a severely spalled bearing or one with several spalled areas, would be more readily detectable than a bearing with a lightly spalled defect.

Examples of the degraded parts used in the airborne tests are shown in Figures 6-1 through 6-10. Figures 6-1 and 6-2 are examples of artificially degraded bearings and Figure 6-7 shows a naturally degraded part. All the parts shown were detected at least once by the AIDAPS during airborne testing.

A summary of the fault detection results by aircraft component is presented in paragraphs 6.2.2 and 6.2.3.

### 6.2.1 Flight Safety Qualification Tests

Flight safety qualification tests were conducted at Ft. Rucker to determine that no interference existed that would affect either AIDAPS or aircraft performance. These tests were made using both Hybrid I and Hybrid II configurations of airborne segment hardware. Test results showed that no interference existed.

### 6.2.2 Implant Tests--Performance Test Period January Through April 1976

Implant tests were conducted on four UH-1H helicopters to determine

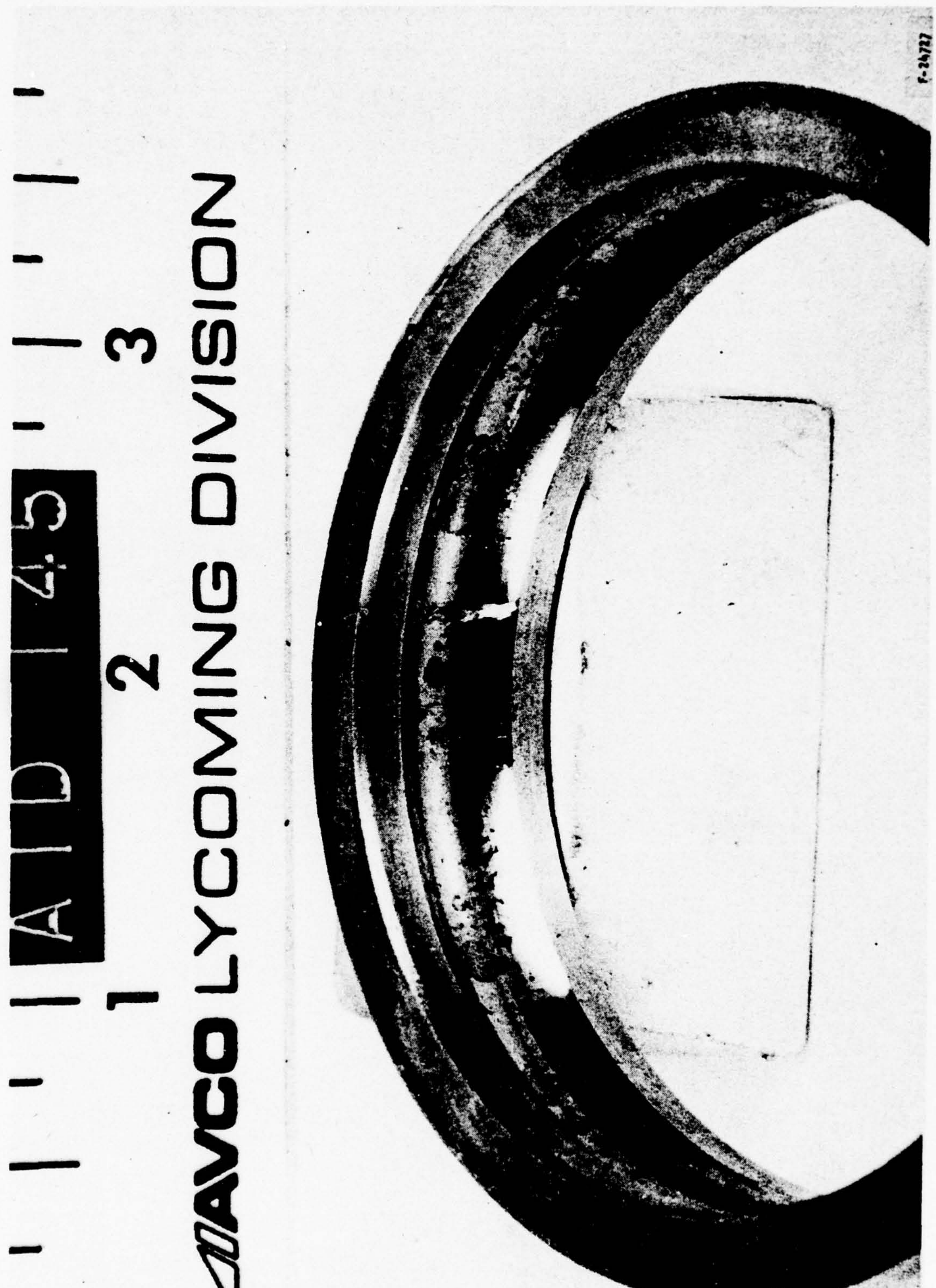



Figure 6-1. AID 145 No. 4 Bearing - Engine (Outer Race Spall)

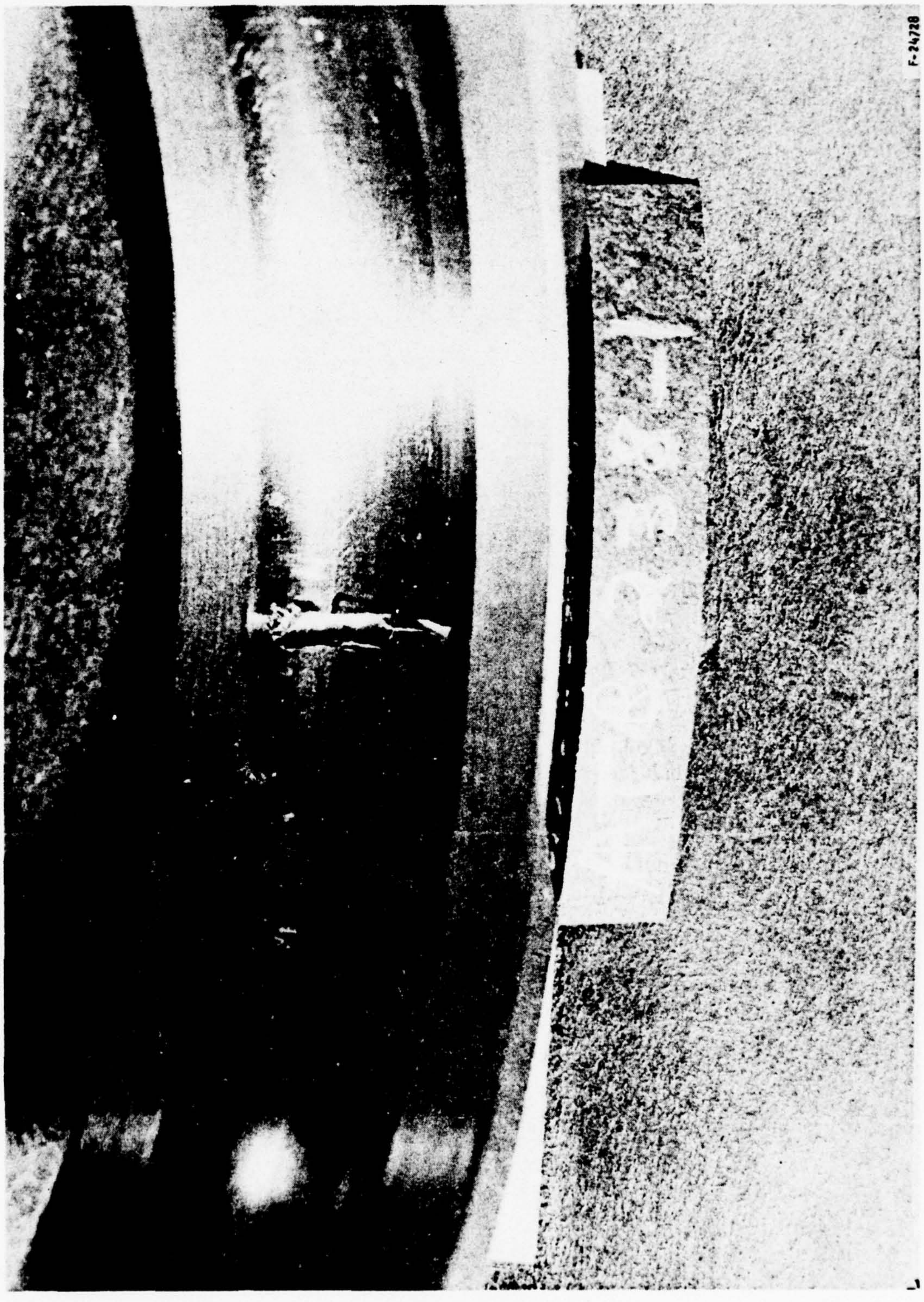



Figure 6-2. ATB 238 No. 1 Bearing - Engine (Outer Race Spall)

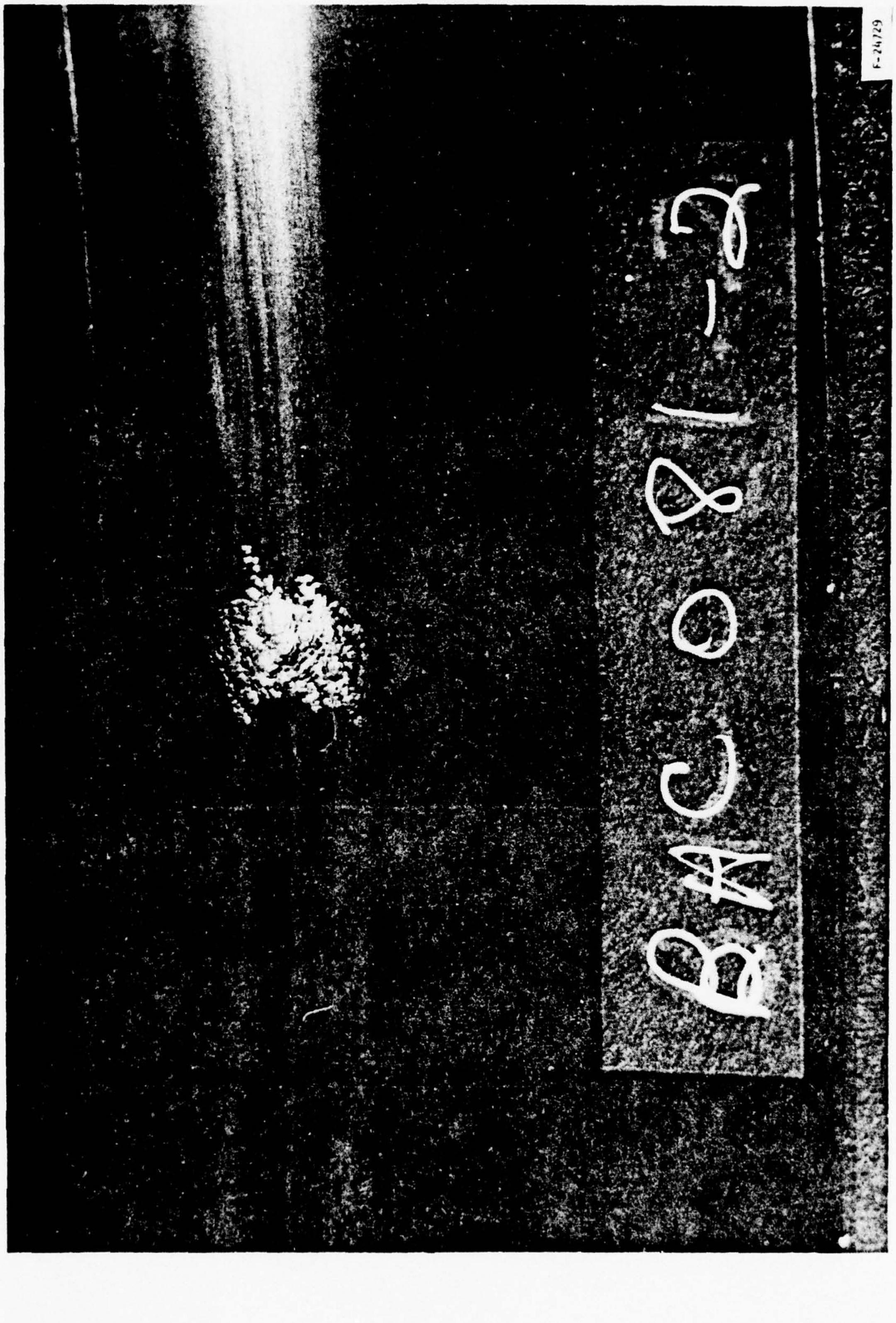



Figure 6-3. BHC 081 Input Drive Quill Bearing - Transmission (Outer Race Spall)

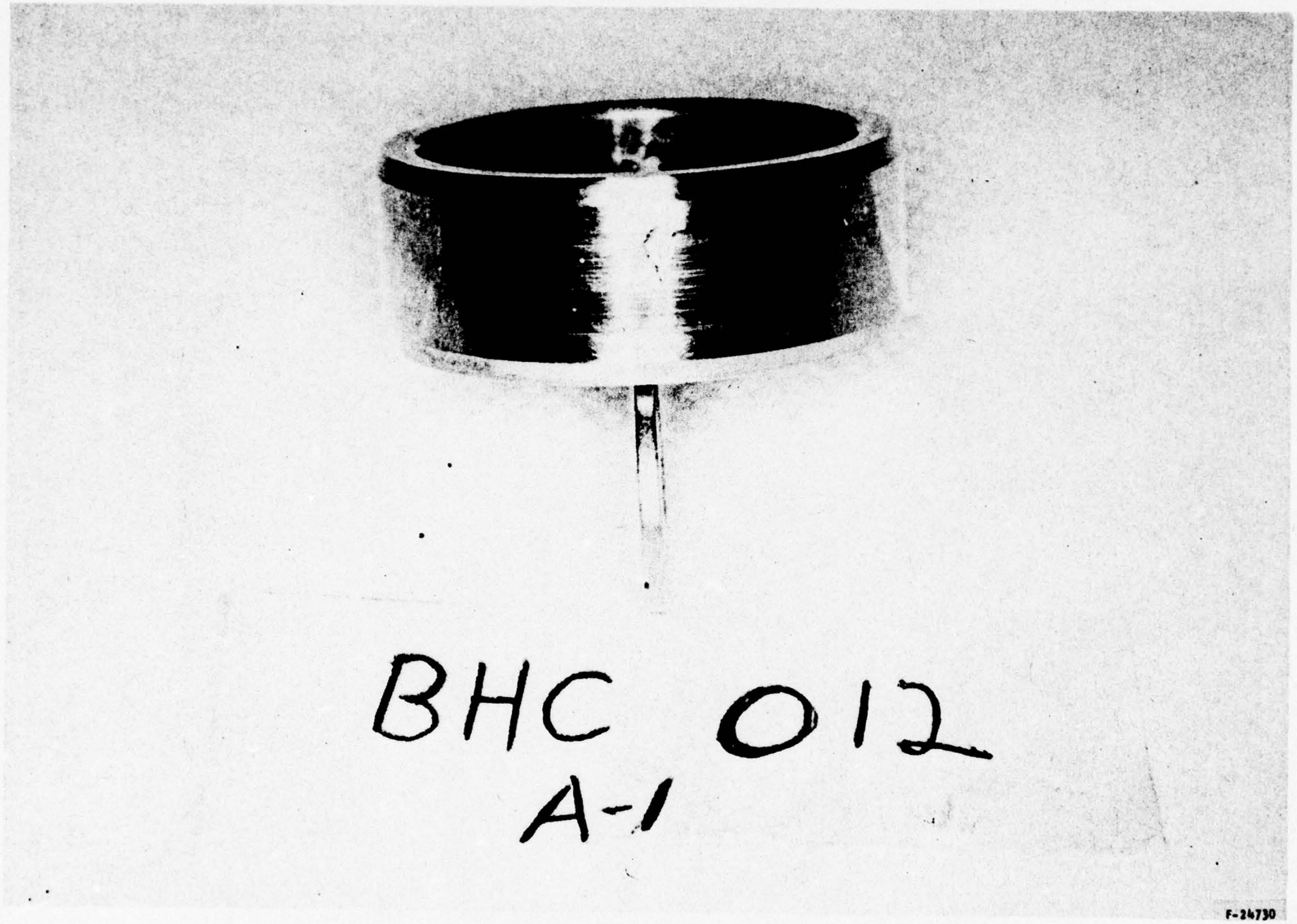



Figure 6-4. BHC 012 Roller Bearing Used in Tail Rotor Output Quill, Transmission, and in $42^0$ Gearbox (Inner Race Spall)

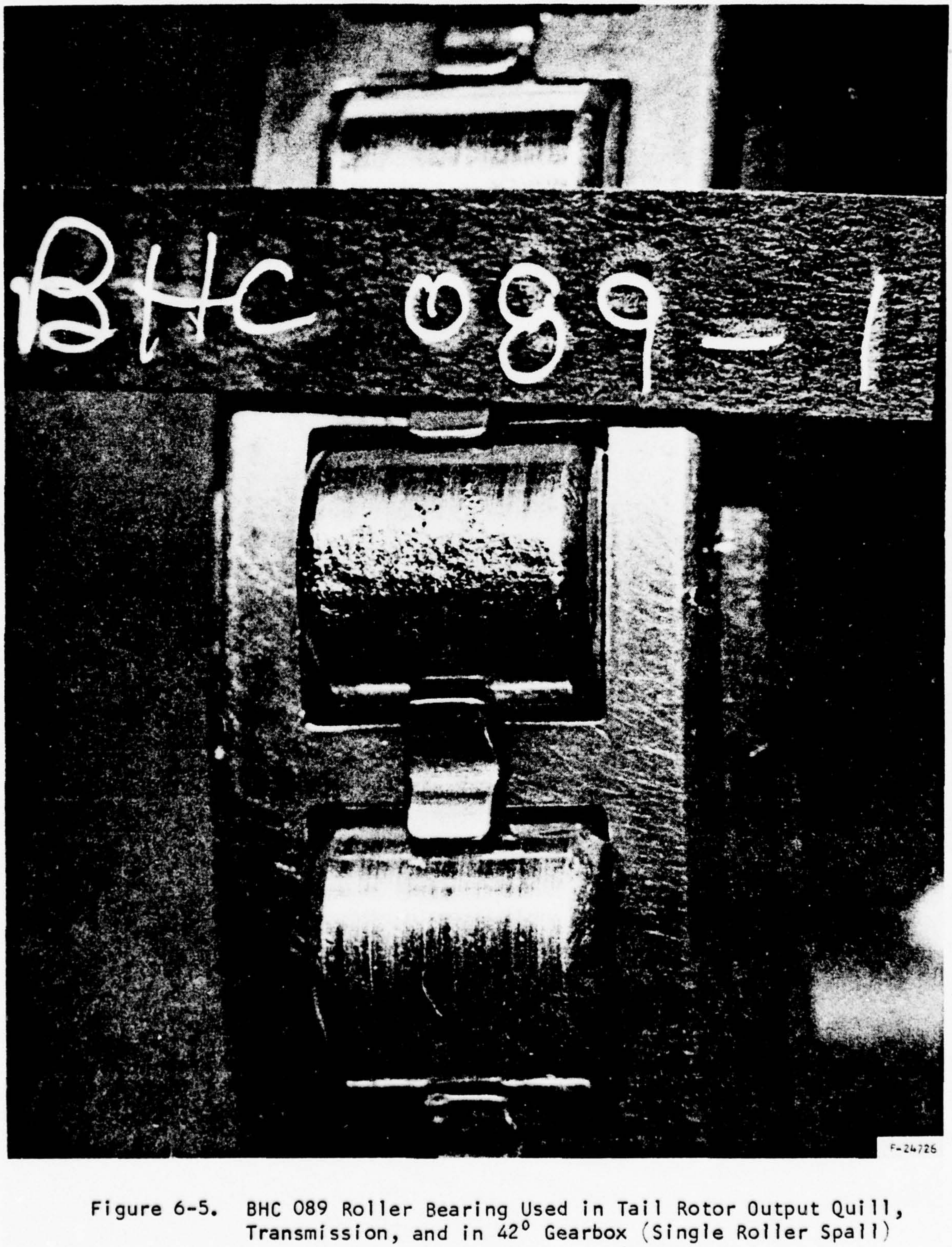



Figure 6-5. BHC 089 Roller Bearing Used in Tail Rotor Output Quill, Transmission, and in 42$^{0}$ Gearbox (Single Roller Spall)

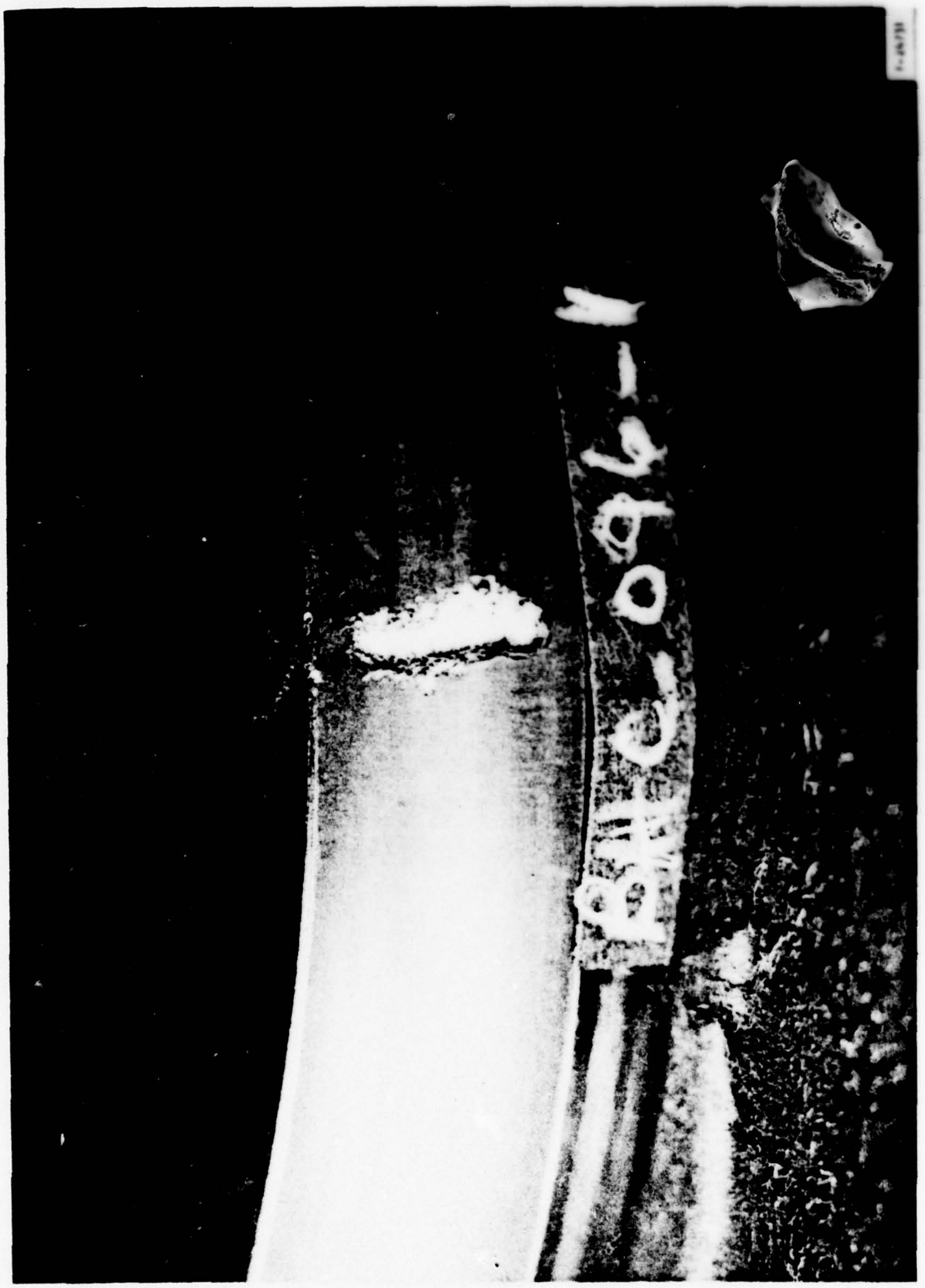



Figure 6-6. BHC 096 Mast Bearing - Transmission (Inner Race Spall)

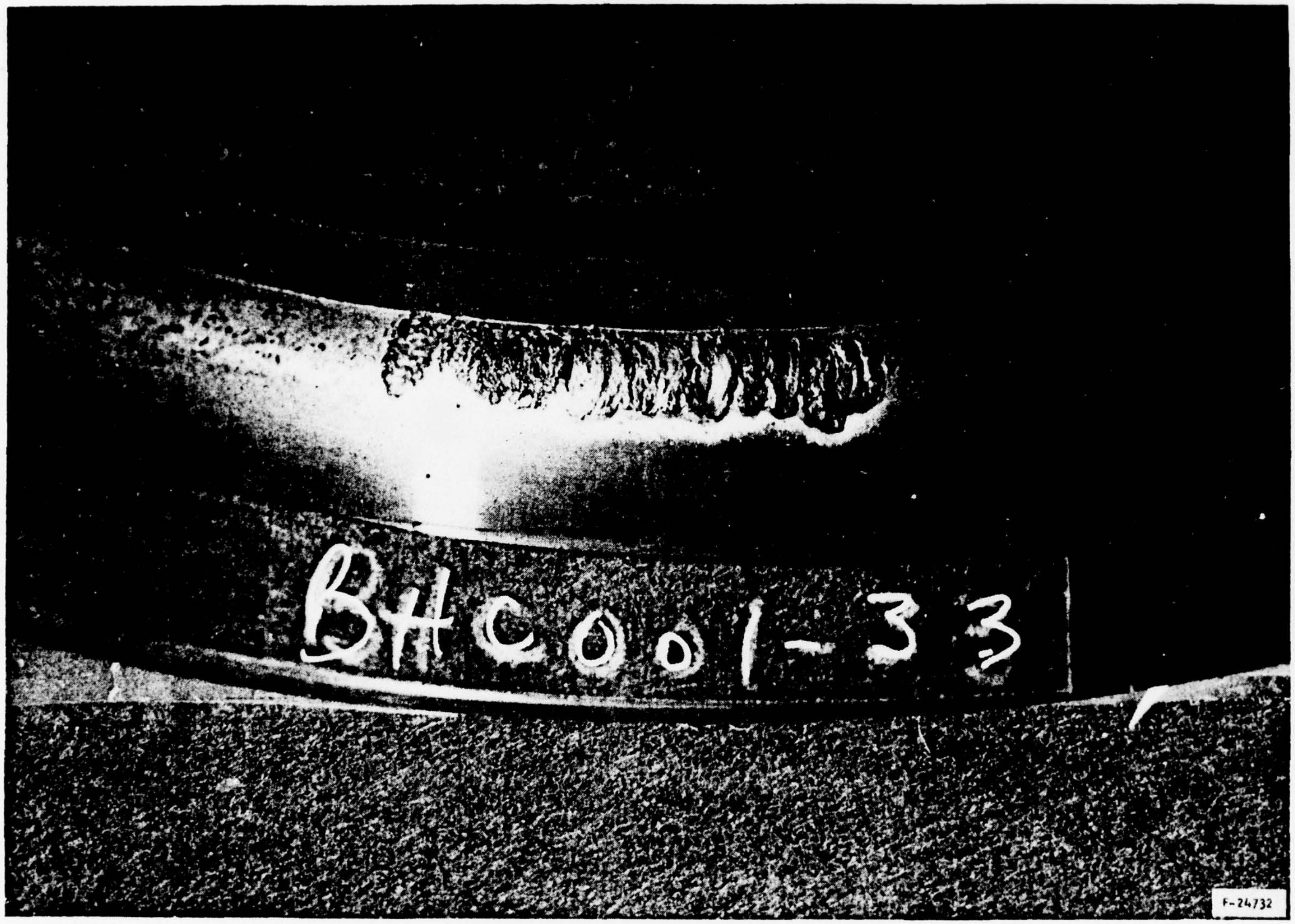



Figure 6-7. BHC 001 Ball Bearing Used in Tail Rotor Output Quill, Transmission, and in $42^0$ and $90^0$ Gearboxes (Outer Race Spall)

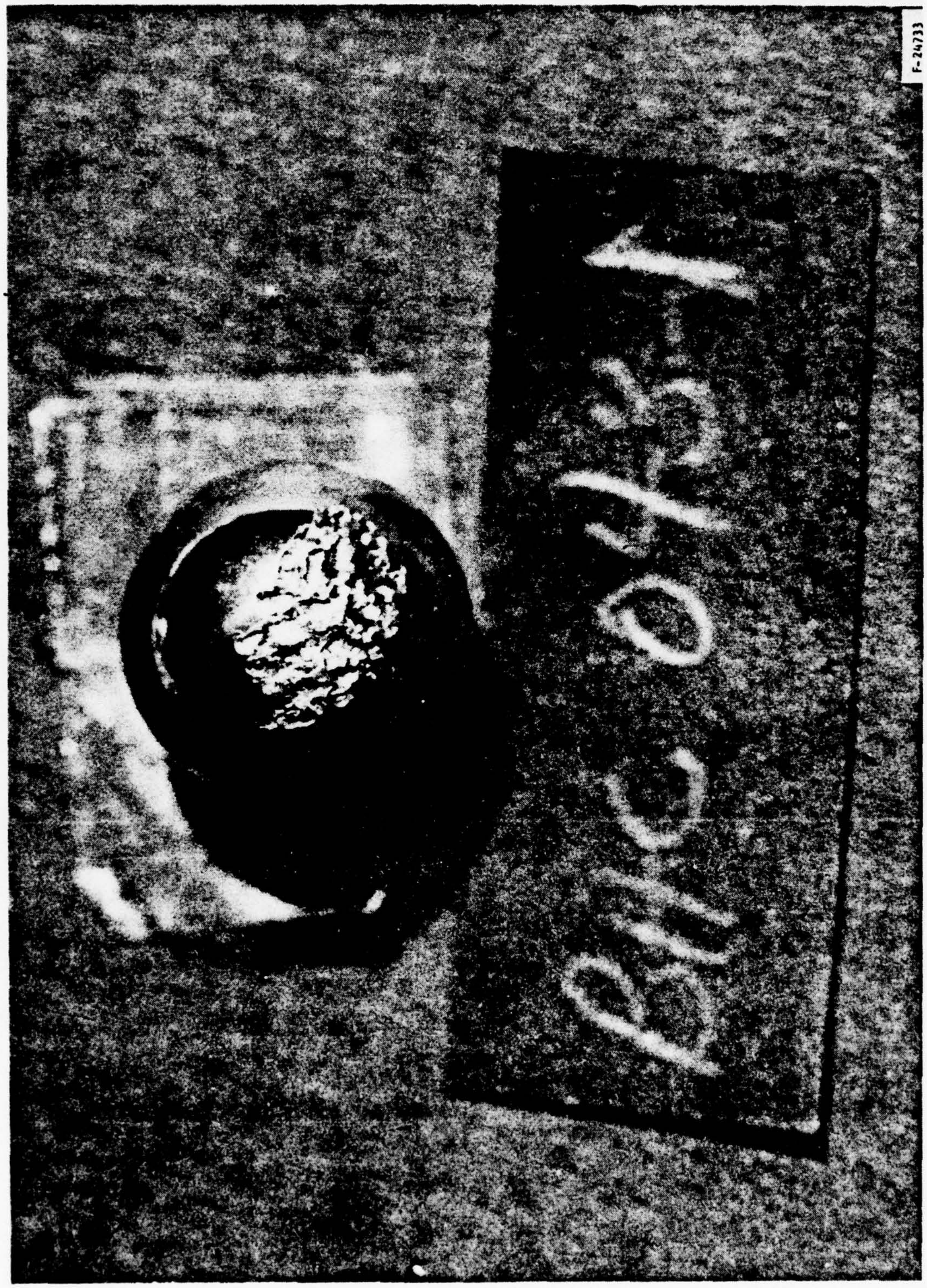


Figure 6-8. BHC 043 Ball Bearing Used in Tail Rotor Output Quill, Transmission, and in 42° and 90° Gearboxes (Single Ball Spall)

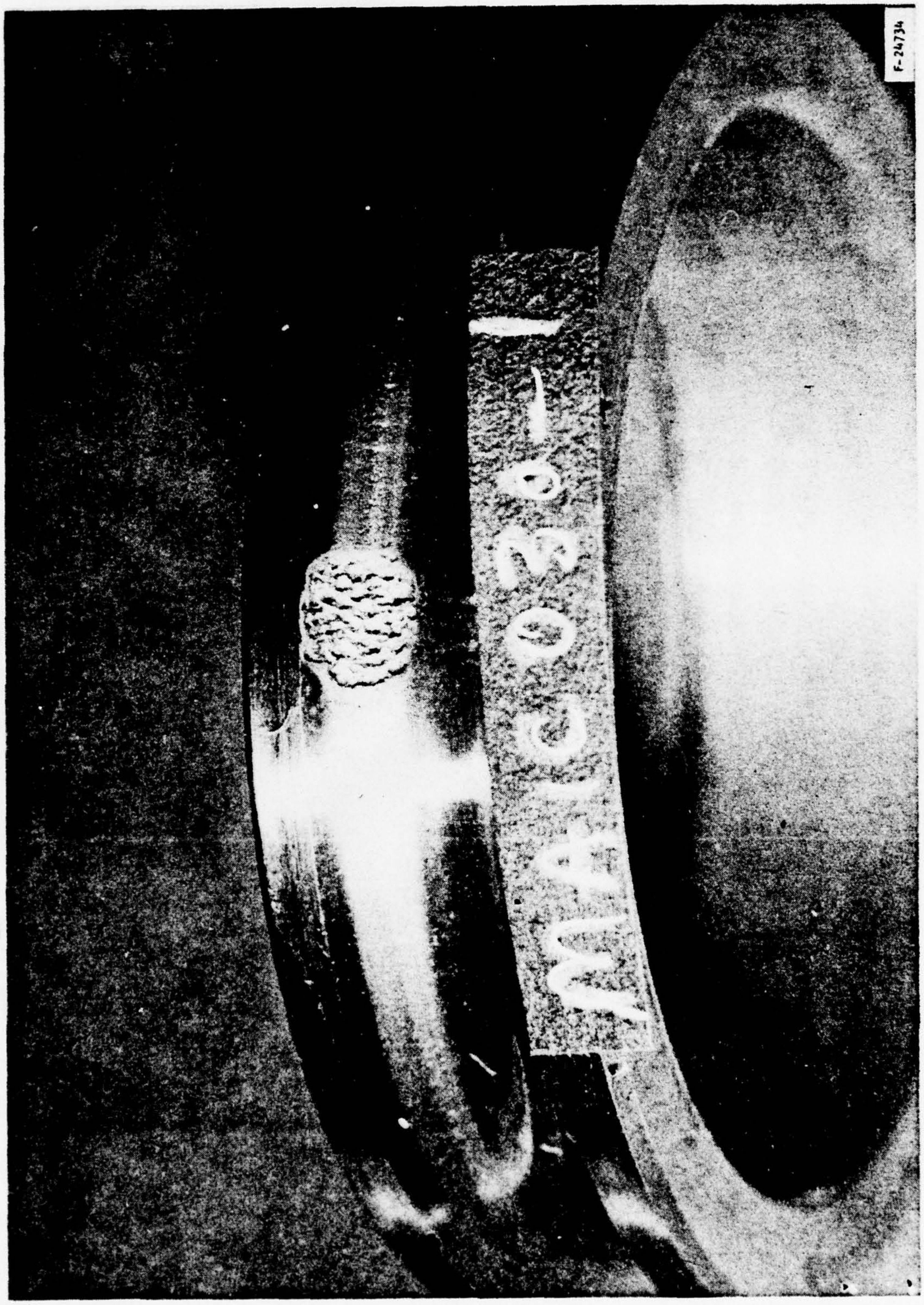
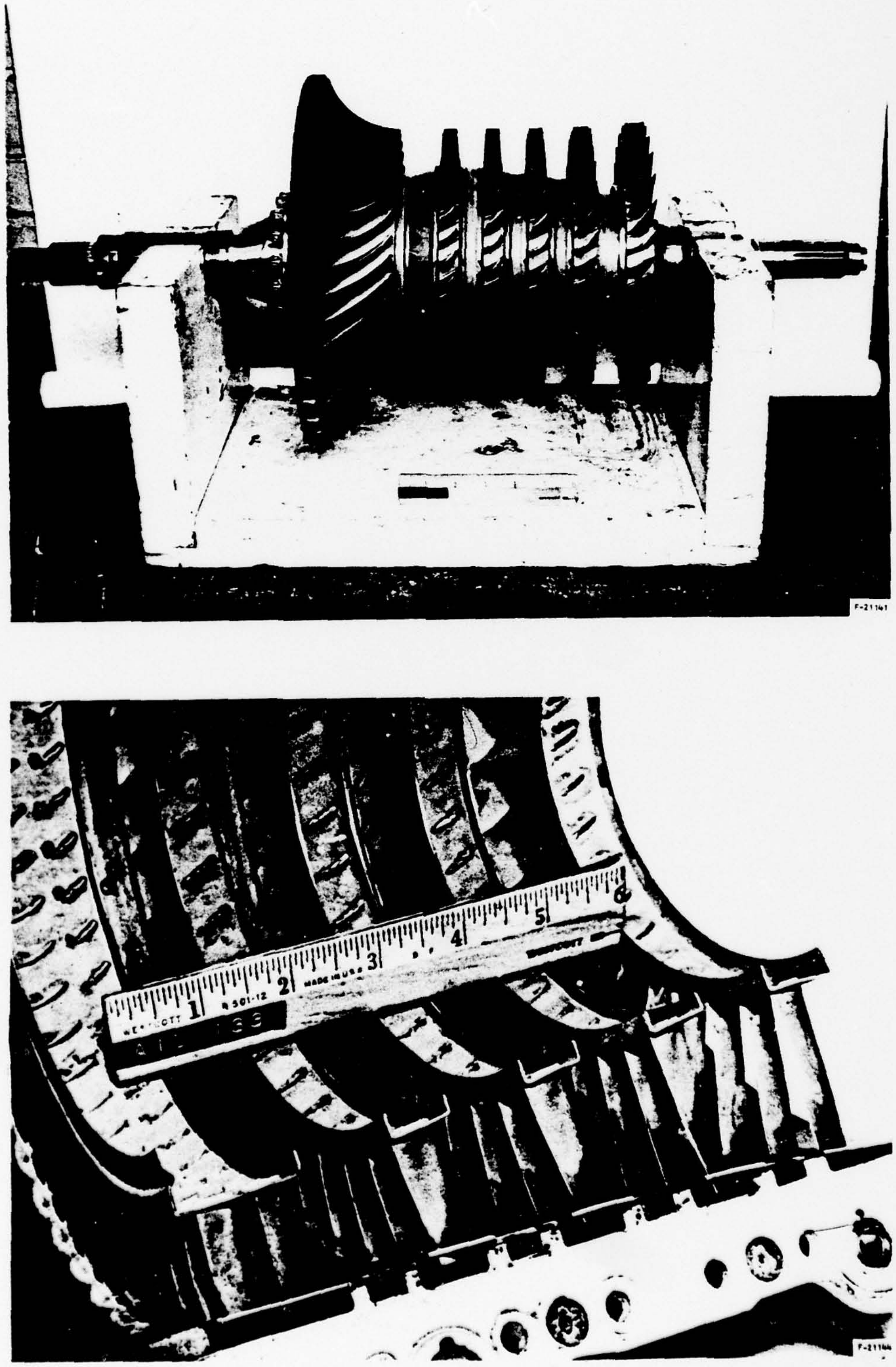



Figure 6-9. MAIC030 Ball Bearing Used in 90° Gearbox (Inner Race Spall)



F-24736

Figure 6-10. AID 132, 133 Eroded Compressor Section

system performance in terms of fault-detection capability. These tests allowed fault-detection logic limits to be established and system capabilities to be revised as found necessary. Reference 8 lists the results of each test and the revisions made at the end of the performance test, in addition to detailed analysis results.

The implant fault detection tests demonstrated the capability of the system to detect various types of faulty parts and components. A total of 29 different implants was tested in various aircraft and components. Eighty-nine percent of these were clearly detectable, with the remainder being marginally detectable.

The detection scores experienced for each LRU during this period were as follows:

| | Percent |
|---|---|
| Engine - gas path | 84* |
| Engine - vibration | 67 |
| Transmission | 76 |
| $90^{\circ}$ gearbox | 100 |
| $42^{\circ}$ gearbox | 100 |

*Including two gas producer turbine implants which were tested in order to evaluate their detectability. One of these produces parametric changes near the prototype limit; the other below the limit. One-hundred percent of the other implants previously adjudged detectable were detected.

### 6.2.3 Implant Tests--Demonstration Test Period May Through October 1976

The AIDAPS prototype demonstration test was designed to measure how well the prototype system implements the diagnostic logic, which was developed during the data collection and analysis phases of the program. In order to minimize test costs and time, a simulation approach was used. In this test, as in the performance test, component (aircraft LRU) degradations were simulated by installing one or more degraded parts (e.g., bearings, gears, turbine wheels, compressor rotors, etc.) into an otherwise new (or newly overhauled) component.

[illegible]

[illegible]

designed to simulate real conditions as closely as practicable, nevertheless, it is impossible to represent the true degradation condition in all respects and some aspects of the simulation may, of themselves, introduce masking effects on the measurement of system fault-detection and false-alarm rates.

Four important simulation technique factors to be considered are as follows:

1. Both naturally and artificially degraded parts were used in the simulation.

2. The implantation of one degraded part among the remaining like-new parts was supposed to simulate a gradually degraded component.

3. The component must be disassembled in order to implant the part, thus modifying the "build" and probably the "signature" of the component.

4. Repeated disassembly in order to test many different parts required excessive handling of instrumentation (sensors) causing unrealistic instrumentation reliability and performance degradation problems.

These factors were considered during the evaluation of the demonstration test results.

During the demonstration test period, 156 different degraded parts and 60 components were tested on four aircraft. The test results were carefully analyzed in order to remove invalid tests due to incorrect part installations, procedural problems, and system hardware failures. The resulting fault detection score for implanted, degraded parts is as follows:

| Component | Fault Detection Percentage |
|---|---|
| Engine gas path | 78 |
| Engine vibration | 48 |
| Transmission | 85 |
| Hanger bearing | 85 |
| [illegible] gearbox | 100 |
| [illegible] gearbox | 44 |

Overall score for all components = [illegible] percent

False Alarm Rate During Demonstration Test Period

The false alarm rate experienced during this period was 2.6 percent. Subsequent testing indicated that many of the false alarm events may be due to the maintenance actions required in the implantation process. As a result, a more realistic test was designed and implemented at the end of the demonstration test period. In this test, one AIDAPS-equipped aircraft was restored to baseline condition. The aircraft was then flown on normal missions logging approximately four flight hours (two flights) per day. As of this writing, approximately 80 hr of flight have been flown with no false alarms. From this test it is concluded that the anticipated false alarm rate is indeed low and that maintenance action probably made a significant contribution to the false alarm rate experienced during implant testing.

## 6.3 SYSTEM RELIABILITY

AIDAPS system operation started at Ft. Rucker early in January, 1976. By early November the four test aircraft had been flown approximately 754 system test hours. During this period an accurate record was kept of all system hardware failures, and trouble reports were prepared for each failure. Appendix C is a summary of these reports showing Trouble Report number, date of occurrence, aircraft number, failed item and the nature of the failure. From this data an achieved mean-time-between-failure (MTBF) number was calculated for each system component. Also, since a number of correctable design-related failures occurred, a projected MTBF number was calculated assuming the design deficiencies were corrected. These calculations are contained in Appendix D. The calculated MTBF numbers are shown in Table 6-1. The projected MTBF of 945 hr for the Hybrid I airborne segment indicates that for an average flying time of 40 hr per month, one AIDAPS failure will occur every 23 months.

As a result of reliability and trouble report analyses, the following recommendations are made for improving the system reliability:

Aircraft wiring and installation-related

1. Compressor discharge temperature thermocouple wire

   Ten trouble reports were written related to the CDT thermocouple wire or the termination lugs on the wire at the transducers in the engine compartment. Many were wires broken including some where the wire broke inside the sheath.

   Conclusion: The wire runs to the CDT transducer and the termination method at the transducers are inadequate for the job.

   Recommendation: Change the transducer from lug connections [illegible]

TABLE 6-1

AIDAPS MTBF NUMBERS

| System Component | Objective | MTBF Achieved | MTBF Projected |
|---|---|---|---|
| Digital data recorder | - | 293 (1) | 15,000 |
| Data acquisition unit | - | 299 (2) | 1,194 |
| Computer memory unit | - | >377 (3) | 4,545 |
| Sensors (average per sensor) | 2,000 | 1,508 (4) | 3,975 |
| Diagnostic analyzer | - | 293 (5) | 880 |
| Aircraft wiring and installation-related | - | 3,132 | 6,264 |
| Airborne segment-Hybrid I (DAU, CMU) | 1,000 | >167 | 945 |
| Airborne segment-Hybrid II (DAU, DDR) | 1,000 | 148 | 1,106 |

NOTES:

(1) Digital data recorder. All failures (three) occurred on one recorder, SN 85-D2. The other unit was trouble-free. Projected MTBF derived from manufacturer's unit specifications.

(2) Data acquisition unit. Three trouble reports relate to failure of multiplexer input for the vibration channels. After a fix was developed and installed (~9/22, TR29288) no further failures have occurred.

(3) CMU - No failures in 377 operating hr. MTBF not definable. Projected MTBF is engineering/reliability prediction.

(4) Sensors - The sensor group contained three marginal units. (1) compressor inlet temperature probe, (2) compressor discharge temperature, and (3) an accelerometer. See reliability recommendations below for further discussion.

(5) Diagnostic analyzer. Two diagnostic analyzer problems were core memory design problems corrected by the vendor (TR[illegible]).

2. Torque pressure bulkhead feedthroughs

When designed, the torque pressure transducer installation used different size fittings and lines. When the installations were made, not all parts were available and the same size bulkhead feedthroughs were used with a reducer. Twice during the testing these lines have been connected backwards damaging the pressure transducer.

Recommendations: Obtain the correct parts and change the aircraft installation to prevent reoccurrence of problem.

3. Accelerometer mounting

Present indications are that improper mounting of accelerometers can result in major changes in accelerometer frequency response, mainly at the high frequencies. See TR29084 and vendor failure analysis.

Recommendations: Re-inspect accelerometer mounting brackets, and the individual accelerometer mounting surface for smoothness and absence of burrs and ridges. Modify installation procedure to clarify method of mounting and to incorporate inspection criteria.

<u>Sensors</u>

1. Accelerometer 6222M26

Eleven trouble reports have been written against this accelerometer--four shorted, one open-circuited, a number for frequency response changes (one confirmed). The vendor's report confirms the shorts, but ascribes the frequency response changes to mounting problem (discussed above). AIDAPS system uses 13 accelerometers (three are 6233's, a different, but related part).

Recommendations: For continuation of the AIDAPS program all accelerometers should be returned to the vendor (or some other testing facility) for retest. These should include: (1) frequency response to 50 kHz for the 6222M26, and to 5 kHz for the 6233), (2) resistance, (3) isolation to case, (4) capacitance, (5) capacitance unbalance, and (6) charge sensitivity. This testing should provide a proper basis from which to proceed on a new program. Also, the vendor should press on his analysis of the shorting problem.

For a new program, work should continue with the present vendor to improve his product, and, in addition, other sources should be evaluated.

2. CDT thermocouple probe HAC15753

Four failures, all open elements where the wire exits from the insulating material (open-probe element). Vendor suggested high vibration as the cause and recommended an encapsulated unit. Time constant is not a problem for this probe.

Recommendations: Develop new probe for this application, sealed and/or encapsulated, with a connector rather than lugs (see aircraft wiring recommendations above).

3. CIT resistance probe 56BP178

Six failures--three open, one shorted to case, one shifting value, indicate that this probe is not physically adequate for the required application. This type of probe has not been a problem on past programs.

Recommendations: Work with the present vendor to improve his product, and/or find an alternate source for this probe.

It is further recommended that these improvements be incorporated into AIDAPS before proceeding with further testing or capability development using the present prototype hardware.

## 6.4 RECOMMENDATIONS

As a result of the performance and demonstration tests, several recommendations were made (see Reference 8). These recommendations can be grouped into four major areas as follows:

- Correct system hardware deficiencies to improve reliability and system performance.
- Improve maintenance personnel/system interface by revising operating procedures and maintenance message content and format.
- Incorporate prognosis to enable prediction (and determination of) limiting part condition for removal from aircraft use.
- Revise fault-detection logic limits to conform with signature changes corresponding to established limiting part conditions.

It is further recommended that these problem areas be corrected prior to any future service tests to verify the capability of AIDAPS prototype hardware to provide on-condition maintenance of helicopter components.

## 7. CONTRACTUAL DATA ITEMS

The following is a complete listing of the contractual data items submitted during the period of the AIDAPS Contract, each item identified by its CDRL Sequence, Data Item Description (DID), and AiResearch Report, where applicable:

| CDRL | Description |
|---|---|
| - A001 | Contract Funds Status Report, DI-F-6004, quarterly during life of Contract |
| - A002 | Procurement Information Report, DI-F-6001, quarterly during life of Contract |
| - A004 | System Specifications, DI-E-1104A, AiResearch Report No. 76-12464 |
| - A005 | Cost and Performance Reports, DI-F-1208, monthly during life of Contract |
| - A006 | Cost Planning and Appraisal Chart (CPA), DI-F-1202, monthly during life of Contract |
| - A008 | Aperture Cards, DI-E-1112 |
| - A009 | Engineering Data (Categories E & F) for procurement Items and Item Components, DI-E-6112 |
| - A010 | Engineering Data (Category G) for Installation, Test Configuration, DI-E-6113; Gas Path and Vibration Data Collection Systems, Set of drawings and supplementary engineering data |
| - A012 | Engineering Data (Category G) for Installation, Final Configuration, DI-E-6113; Set of 36 drawings and supplementary engineering data |
| - A013 | Software for other than Business-Oriented Computer Programs/ Systems, DI-E-1125, titles as follows: |

Gas-Path Quick Look Processing Computer Program

Broadband Amplitude Analysis Computer Program

| CDRL (Cont) | Description |
|---|---|
| - A013 (Cont) | Vibration Data Processing and Reduction Computer Program |
| | Gas-Path Sensor Calibration History Computer Program |
| | Likelihood Analysis Computer Program |
| | Gas Path Data Processing and Engineering Units Conversion Computer Program Documentation |
| | Gas Path Data Analysis Computer Program |
| | Operating Instructions for AIDAPS Gas Path Programs |
| | Program MAINL (Mainline Program, Gas Path, DA) |
| | Subroutine DIAGN (System Diagnostics, Gas Path, DA) |
| | Subroutine CREDT (Credibility Checks, Gas Path, DA) |
| | Subroutine IGVCK (Inlet Guide Vane Check, Gas Path, DA) |
| | Subroutine DEPVC (Dependent Variable Corrections, Gas Path, DA) |
| | Subroutine VARDF (Variable Differentials, Gas Path, DA) |
| | Subroutine DELTA (Calculation of Variable Differentials, Gas Path, DA) |
| | Subroutine MODEL (Thermodynamic Model Calculations, Gas Path, DA) |
| | Subroutine PMESS (Performance Messages, Gas Path, DA) |
| | Subroutine CONFL (Confidence Index, Gas Path, DA) |
| | Subroutine BSLNE (Baseline Update, Gas Path, DA) |
| | Subroutine LSFIT (Least Squares Fit, Gas Path, DA) |
| | Subroutine GPROG (Prognosis, Gas Path, DA) |
| | Subroutine PCOMP (Error Compensation) Gas Path, DA) |
| | Gas Path Flight Logic |
| | AIDAPS Diagnostic Analyzer (DA) Computer Program |
| | AIDAPS Airborne Computer Memory Unit (CMU) Computer Program |

| CDRL (Cont) | Description |
|---|---|
| - A013 (Cont) | DAU Interrupt Routine Computer Program |
| | AIDAPS Vibration Monitor Subroutine Computer Program |
| | Mechanical-Functional Routing Computer Program |
| | Message Handler Routine Computer Program |
| | Built-In Test Routine Computer Program |
| | Advance/Display Routine Computer Program |
| | DTU Store Subroutine Computer |
| | Subroutine EXPRO (Execute Processing) Computer Program |
| | Main Program for DA Computer |
| - A014 | Subsystem Test Plan, DI-T-1903, AiResearch Report No. 73-9492, including Revisions 1 through 19 |
| - A015 | • Coordinated Test Plan, DI-T-1900, AiResearch Report No. 75-11738, Final Approved document |
| - A016 | Test and Demonstration Report, DI-T-1906, AiResearch Report No. 76-12840, and Revision 1 |
| - A017 | Technical Report - Accumulated Data, DI-S-1800, AiResearch Report No. 76-13186 |
| - A018 | Final Technical Report, DI-S-1800, AiResearch Report No. 76-13354 |
| - A019 | Interim Technical Report, DI-S-1800, a series of six 6-month interim reports during life of program as follows: |
| | First Interim Techical Report, AiResearch Report No. 74-10011 |
| | Second Interim Technical Report, AiResearch Report No. 74-10718 |
| | Third Interim Technical Report, AiResearch Report No. 75-11232 |
| | Fourth Interim Technical Report, AiResearch Report No. 75-11717 |
| | Fifth Interim Technical Report, AiResearch Report No. 76-12372 |

| CDRL (Cont) | Description |
|---|---|
| - A019 (Cont) | Addendum I to Fifth Interim Technical Report, AiResearch Report No. 76-12372, Addendum I<br><br>Sixth Interim Technical Report, AiResearch Report No. 76-13172 |
| - A020 | Letter Progress Technical Reports, DI-S-1800, monthly reports submitted throughout life of Contract, AiResearch Report No. 75-11285 (1) through (19). (No numbers assigned to reports prior to March, 1975). |
| - A021 | Training and Equipment Plan, DI-H-6131 (RFP response) |
| - A022 | Reliability Reports, DI-R-1731, quarterly reports submitted for periods through March 1975 (Requirement deleted by Contract Mod. P00020). |
| - A023 | Maintainability Reports, DI-R-1741, quarterly reports submitted for periods through March, 1975. (Requirement deleted by Contract Mod. P00020). |
| - A024 | Calibration Requirements Summary, DI-L-1401, AiResearch Report No. 73-9523 |

- In addition to the above, the following items did not appear on the CDRL but were submitted as required elsewhere in the Contract:

Error Source Study, AiResearch Report No. 76-12696

Evaluation of Vibration Analysis Techniques, AiResearch Report No. 74-9934

Detailed Plan of Contract Performance

Detail Plan of Performance for Phase VIII, Second Generation AIDAPS

## REFERENCES


1. AIDAPS Staff, First Interim Technical Report, AiResearch Report No. 74-10011, December 30, 1973. (Covers period from 27 June to 31 December 1973).

2. AIDAPS Staff, Second Interim Technical Report, AiResearch Report No. 74-10718, November 8, 1974. (Covers period from 1 January to 30 June 1974).

3. AIDAPS Staff, Third Interim Technical Report, AiResearch Report No. 75-11232, April 1, 1975. (Covers period from 1 July to 31 December 1974).

4. AIDAPS Staff, Fourth Interim Technical Report, AiResearch Report No. 75-11717, August 1, 1975. (Covers period from 1 January to 30 June 1975).

5. AIDAPS Staff, Fifth Interim Technical Report, AiResearch Report No. 76-12372, February 27, 1976. (Covers period from 1 July to 31 December 1975).

6. AIDAPS Staff, Addendum 1 to Fifth Interim Technical Report, AiResearch Report No. 76-12372, Addendum 1, 12 August 1976. (Covers period from 1 July to 31 December 1975).

7. AIDAPS Staff, Sixth Interim Technical Report, AiResearch Report No. 76-13172, 8 October 1976. (Covers period from 1 January to 30 June 1976).

8. AIDAPS Staff, Phase V Test and Demonstration Report, AiResearch Document No. 76-12840, 1 September 1976.

9. AIDAPS Staff, Supplement 1 to Phase V Test and Demonstration Report, AiResearch Document No. 76-12840, Revision 1, 1 December 1976.

10. Minnear, J., Subsystem Test Plan, AiResearch Document No. 73-9492, Revision 19, 21 July 1975.

11. AIDAPS Staff, Technical Report - Accumulated Data, AiResearch Document No. 76-13186, November 4, 1976.

12. Chang, J. D., Elder, C., Harris, J., and Lau, J., Software for other than Business-Oriented Computer Programs/Systems, CDRL Data Item No. A013, group of 34 AiResearch documents.

13. Chang, J. D., Evaluation of Vibration Analysis Techniques, AiResearch Report No. 74-9934, 7 February 1974.

APPENDIX A

AIDAPS COST/BENEFIT COMPUTER MODEL (AIDCST)

## AIDAPS COST/BENEFIT COMPUTER MODEL (AIDCST)

The AIDCST computer program has been developed for and applied to estimating the cost benefits that can be expected as a result of the use of AIDAPS equipment on Army helicopters. The program has been written in FORTRAN V and used on the AiResearch Sperry Rand 1100 computer.

The program involves computation of the following cost elements:

- Program costs associated with design, development, test and evaluation (DDT&E); investment for the procurement and installation and operation of AIDAPS equipment.
- Program cost savings for
  - Reduced inspection time for daily, intermediate, and periodic inspection
  - Reduced troubleshooting time associated with the diagnosis of failures
  - Reduced unwarranted removals of equipment with indications of failure or potential failure
  - Reduced bench check associated with unwarranted removals
  - Reduction or elimination of scheduled removals of monitored subsystem components
  - Reduction in shipping and packaging costs associated with both unwarranted and scheduled removals
  - Reduction in initial spares costs associated with the reduced frequency of removals
  - Reduction in overhaul costs due to the reduction due to reduced removals
  - Reduction in overhaul cost for warranted removals due to reduction in secondary damage
  - The effective decrease in cost in terms of equivalent aircraft cost due to decreased downtime, increased availability of aircraft
  - Reductions in in-flight incidents causing actual or potential replacement of lost or damaged aircraft and incomplete missions causing reflights (false alarms are included)
  - Reductions due to change in distributions between aircraft losses, major and minor accidents and incomplete mission landings

The following sections present a brief description of the computations performed in the model, input data requirements and output data.

## Model Description

The computer model is a series of elements required to initialize input and computed quantities, read data, operate on the data and print the results of computations. These functions are described for each program element.

| | |
|---|---|
| MAIN program: | An executive element that causes the input data to be read and calls all program subroutines in proper order. This element is arranged for calling multiple cases. |
| Block data: | An element that initializes all variables to zero prior to reading the input data for the first case. |
| Subroutine INSPKT: | Computes and aggregates downtime and manhour reductions associated with the three different types of inspections for all components included in the analysis. |
| Subroutine TRBLSH: | Computes and aggregates downtime and manhours associated with troubleshooting for all components. |
| Subroutine UNWAR: | Computes and aggregates the downtime associated with unwarranted removal actions and related bench testing, accumulates the cost of utilization of new and overhauled spares associated with some unwarranted removals, computes the shipping and packaging costs for spares, and computes initial spares requirements for given number of months of inventory requirement. |
| Subroutine SCHED: | Similar to UNWAR except for items removed in scheduled maintenance actions. Bench checking is not required in this calculation. |
| Subroutine HZARD: | Computes and aggregates the reduction in number of in-flight failures causing aborts, computes the distribution of in-flight failures among aircraft losses, major and minor accidents and various types of abort landings, includes false alarms in abort landings. |
| Subroutine EFFECT: | Computes the increase in fleet availability due to decreased downtime. |

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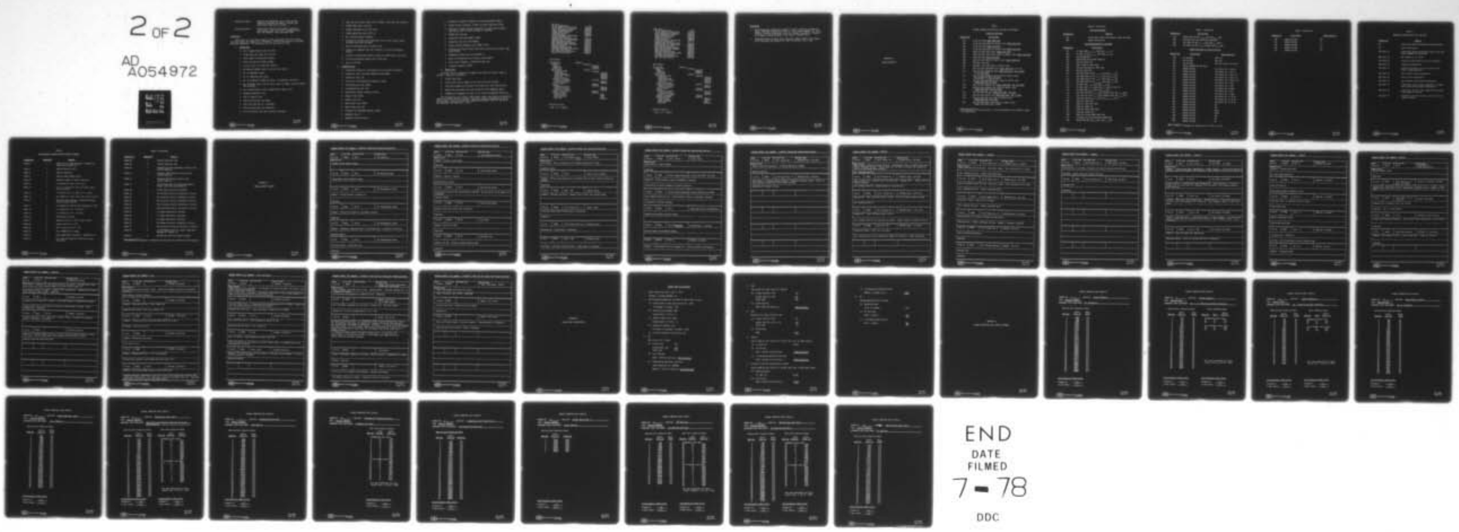

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Subroutine COSTS: Computes and aggregates costs in the various categories required for output, develops cash flow history required for output.

Subroutine PRINT: Causes the printing of the output including a cost summary, cash flow history component group data and component input and output data.

Input Data

Input data is of two types; namely, (1) that associated with the aircraft, the fleet and/or the common to all components, and (2) data related to individual monitored components. The following lists the two types of data:

1. System Data

- No. of operationally ready aircraft.
- Flight hours per month per aircraft.
- Total number of operational months.
- Fleet availability without AIDAPS.
- Aircraft cost in current year dollars.
- Maintenance manhour costs in current year dollars
- No. of component groups.
- No. of components per group.
- No. of inspection categories (daily, intermediate, periodic).
- No. of hazard types (aircraft loss, major accidents, minor accidents and landings).
- No. of mainteanance levels (organization, depot, etc.).
- Economic escalation rate.
- Cost of capital rate.
- Start and stop year for DDT&E.
- Start and stop year for investment.
- Start and stop year for operations.
- No. of different cash flow situations analyzed.

- Cost and cost saving items to be included in the cash flow analyses.
- AIDAPS DDT&E costs, dollars.
- AIDAPS investment cost, $/aircraft.
- AIDAPS operations costs, $/flt hr.
- No. of months spares inventory.
- Fraction of aircraft cost associated with aircraft losses, major accidents and minor accidents.
- Ratio of overhauled parts to parts used.
- Fraction of component cost for shipping to various maintenance levels.
- Fraction of unwarranted removals requiring overhauled or new parts.
- Aircraft maintenance manhours per flight hour.
- Cost of a flight.

2. Component Data

- Inspection frequencies, maintenance actions/flight hour, MA/FH.
- Inspection times, hours/maintenance action, HR/MA.
- Inspection crew size.
- On-aircraft troubleshooting frequency, MA/FH.
- Troubleshooting time, HR/MA.
- Troubleshooting crew size.
- Unwarranted removal frequency, MA/FH.
- Removal time, HR/MA.
- Removal crew size.
- Bench check time, HR/MA.
- Bench check crew size.
- Frequency of scheduled removals, MA/FH.
- Component cost, $
- Component overhaul cost, $

- Fraction of removals shipped to various maintenance levels.
- Flight failure frequency, flight failures/flight hour FF/FH.
- Fraction of flight failures resulting in a particular accident type (loss, major or minor accident or landing).
- AIDAPS test accuracy.
- Inspection time with AIDAPS, HR/MA.
- Inspection crew size with AIDAPS.
- Flight failure frequency with AIDAPS, FF/FH.
- Fraction of flight failures resulting in particular accident types using AIDAPS.
- Component overhaul cost using AIDAPS, $
- Ratio of overhauled parts to parts using AIDAPS.
- False alarm frequency, incidents/flight hour.
- Frequency removals, MA/FH.

3. Output Data

The PRINT routine automatically commands the output of several types of printout. These are as follows:

- System input data.
- Current year dollar summary of all costs and cost savings.
- Cash flow assumptions and cash flow history for input time periods.
- Quantities associated with cost saving for each component group.
- Component-by-component printout of all input and computed quantities.

The accompanying two printouts show typical system input data and 1977 dollar summaries of costs and cost savings. Cash flow, group and component printouts are not shown. The input system and component maintenance data for the UH-IH were obtained from References 1 and 2.

INPUT DATA

| | |
|---|---|
| NO. OF AIRCRAFT(AC) | 500 |
| FLIGHT HOURS PER MONTH(FH/MO) | .2000+02 |
| NO. OF OPERATIONAL MONTHS(MO) | .1200+03 |
| FLEET AVAILABILITY(HR/HR) | .8000+00 |
| FLIGHT DURATION(HR/FLT) | .3700+00 |
| AIRCRAFT COST($/AC) | .2240+07 |
| LABOR RATE($/MH) | .1328+02 |
| NO MONTHS SPARES INVENTORY(MO) | .4000+01 |
| AIDAPS IDENTIFICATION | 0 |
| SHIPPING COST RATIO-FIELD($/$) | .3710-02 |
| SHIPPING COST RATIO-DEPOT($/$) | .8450-02 |
| COST FRACTION-A/C LOSS | .1000+01 |
| COST FRACTION-MAJOR ACCIDENT | .5000+00 |
| COST FRACTION-MINOR ACCIDENTS | .1000+00 |
| FRACTION NEW PARTS | .1000+00 |
| FRACTION OVERHAUL PARTS | .9000+00 |
| FRACTION UNWAR REMOV RORG PARTS | .2000+00 |
| A/C MAINT MANHRS(MH/FH) | .7800+01 |
| A/C INSPECTION MANHRS(MH/FH) | .1744+01 |
| COST OF A FLIGHT($/FLT) | .7860+03 |

COST SUMMARY

| | | | | |
|---|---|---|---|---|
| NET SAVINGS | .2185+09 | | | |
| AIDAPS COST | | -.2054+08 | | |
| DDT&E | | | -.2000+07 | |
| INVESTMENT | | | -.1748+08 | |
| OPERATIONS | | | -.1056+07 | |
| COST SAVINGS | | .2391+09 | | |
| MAINTENANCE LABOR | | | .1390+08 | |
| INSPECTION | | | | .1243+08 |
| TROUBLESHOOTING | | | | .5519+05 |
| UNWARRANTED REMOVAL | | | | .4955+06 |
| BENCH TEST | | | | .5026+06 |
| SCHEDULED REMOVAL | | | | .4199+06 |
| SHIPPING & PACKAGING | | | .1406+08 | |
| UNWARRANTED REMOVAL | | | | .8235+07 |
| SCHEDULED REMOVAL | | | | .5829+07 |
| INITIAL SPARES | | | .3259+08 | |
| OVERHAUL | | | .1536+09 | |
| UNWARRANTED REMOVAL | | | | .4383+08 |
| SCHEDULED REMOVAL | | | | .9320+08 |
| ON CONDITION REMOVAL | | | | .1653+08 |
| AIRCRAFT AVAILABILITY | | | .2496+08 | |
| ACCIDENT REDUCTION | | | .8489+03 | |
| AIRCRAFT LOSS | | | | .0000 |
| MAJOR ACCIDENTS | | | | .0000 |
| MINOR ACCIDENTS | | | | .0000 |
| REFLIGHTS | | | | .8489+03 |

CASH FLOW ANALYSIS

CASE 1 OF 2 CASE(S)

INPUT DATA

| | |
|---|---|
| NO. OF AIRCRAFT(AC) | 2000 |
| FLIGHT HOURS PER MONTH(FH/MO) | .2010+02 |
| NO. OF OPERATIONAL MONTHS(MO) | .1200+03 |
| FLEET AVAILABILITY(HR/HR) | .7550+00 |
| FLIGHT DURATION(HR/FLT) | .3700+00 |
| AIRCRAFT COST($/AC) | .4000+06 |
| LABOR RATE($/MH) | .1328+02 |
| NO MONTHS SPARES INVENTORY(MO) | .4000+01 |
| AIDAPS IDENTIFICATION | 0 |
| SHIPPING COST RATIO-FIELD($/$) | .3710-02 |
| SHIPPING COST RATIO-DEPOT($/$) | .8450-02 |
| COST FRACTION-A/C LOSS | .1000+01 |
| COST FRACTION-MAJOR ACCIDENT | .5000+00 |
| COST FRACTION-MINOR ACCIDENTS | .1000+00 |
| FRACTION NEW PARTS | .1000+00 |
| FRACTION OVERHAUL PARTS | .9000+00 |
| FRACTION UNWAR REMOV RORG PARTS | .2000+00 |
| A/C MAINT MANHRS(MH/FH) | .3900+01 |
| A/C INSPECTION MANHRS(MH/FH) | .1744+01 |
| COST OF A FLIGHT($/FLT) | .2800+03 |

COST SUMMARY

| | | | | |
|---|---|---|---|---|
| NET SAVINGS | .5691+08 | | | |
| AIDAPS COST | | -.7578+08 | | |
| DDT&E | | | -.2000+07 | |
| INVESTMENT | | | -.6992+08 | |
| OPERATIONS | | | -.3859+07 | |
| COST SAVINGS | | .1327+09 | | |
| MAINTENANCE LABOR | | | .2836+08 | |
| INSPECTION | | | | .2497+08 |
| TROUBLESHOOTING | | | | .1109+06 |
| UNWARRANTED REMOVAL | | | | .9960+06 |
| BENCH TEST | | | | .1010+07 |
| SCHEDULED REMOVAL | | | | .1274+07 |
| SHIPPING & PACKAGING | | | .5354+07 | |
| UNWARRANTED REMOVAL | | | | .2955+07 |
| SCHEDULED REMOVAL | | | | .2399+07 |
| INITIAL SPARES | | | .1169+08 | |
| OVERHAUL | | | .5513+08 | |
| UNWARRANTED REMOVAL | | | | .1573+08 |
| SCHEDULED REMOVAL | | | | .3345+08 |
| ON CONDITION REMOVAL | | | | .5945+07 |
| AIRCRAFT AVAILABILITY | | | .1994+08 | |
| ACCIDENT REDUCTION | | | .1222+08 | |
| AIRCRAFT LOSS | | | | .6261+07 |
| MAJOR ACCIDENTS | | | | .6312+07 |
| MINOR ACCIDENTS | | | | .0000 |
| REFLIGHTS | | | | -.3502+06 |

CASH FLOW ANALYSIS

CASE 1 OF 2 CASE(S)

References

1. Maintenance Data Tabulation, Appendix E, Book 4, Northrop Corporation Report, NORT-292, prepared for the U.S. Army Systems Command, St. Louis, Missouri; Concept Formulation Study for Automatic Inspection, Diagnosis and Prognosis System (AIDAPS), Final Report, December 1971.

2. USAAVSCOM Technical Report 75-3, Executive Summary Report-Final Report, UH-1H Assessment and Comparative Fleet Evaluations, April 1975.

# APPENDIX B

# AIDAPS MESSAGES

TABLE 1

MESSAGE NUMBERS DISPLAYED ON CMU AND DA PRINTOUTS

Vibration Messages

| Message No. | DA Printout |
| --- | --- |
| U01 | 42 GB VIB HIGH CHECK CHIP DET* |
| U02 | 42 GB VIB HIGH OVER FLT SAFETY LIMIT CHECK CHIP DET* |
| U03 | 90 GB VIB HIGH CHECK CHIP DET* |
| U04 | 90 GB VIB HIGH OVER FLT SAFETY LIMIT CHECK CHIP DET* |
| U05 | XMSN VIB HIGH INPUT QUILL CHECK CHIP DET* |
| U06 | XMSN VIB HIGH INPUT QUILL OVER FLT SAFETY LIMIT CHECK CHIP DET* |
| U07 | XMSN VIB HIGH CHECK CHIP DET* |
| U08 | XMSN VIB HIGH OVER FLT SAFETY LIMIT CHECK CHIP DET* |
| U09 | HGR BRG 4 VIB HIGH |
| U10 | HGR BRG 4 VIB HIGH OVER FLT SAFETY LIMIT |
| U11 | ENG VIB HIGH COMPRESSOR SECT CHECK CHIP DET*, ACC G/B |
| U12 | ENG VIB HIGH COMBUSTOR SECT CHECK CHIP DET*, NO. 2 SCAV* OR NO. 3/4 SCAV* |
| U13 | ENG VIB HIGH COMBUSTOR SECT OVER FLT SAFETY LIMIT, NO. 2 SCAV* OR NO. 3/4 SCAV* |
| U14 | ENG VIB HIGH COMPRESSOR SECT OVER FLT SAFETY LIMIT CHECK CHIP DET* ACC G/B* |
| U15 | ENG VIB HIGH TURBINE SECT CHECK CHIP DET*, NO. 3/4 SCAV* |
| U16 | ENG VIB HIGH TURBINE SECT OVER FLT SAFETY LIMIT CHECK CHIP DET*, NO. 3/4 SCAV* |
| U17 | ENG VIB HIGH FWD SECT CHECK CHIP DET*, ACC G/B* |
| U18 | ENG VIB HIGH CENTER SECT CHECK CHIP DET*, NO. 2 SCAV* |
| U19 | ENG VIB HIGH FWD SECT OVER FLT SAFETY LIMIT CHECK CHIP DET*, ACC G/B* |
| U20 | ENG VIB HIGH CENTER SECT OVER FLT SAFETY LIMIT CHECK CHIP DET*, NO. 2 SCAV* |

*This message will be printed only if the corresponding chip detector signal indicates chips.

TABLE 1 (Continued)

Gas Path Messages

| Message No. | Meaning |
|---|---|
| E01 | Significant engine performance change detected |
| E02 | Sensor malfunction |

Functional/Mechanical Messages

| Message No. | DA Printout |
|---|---|
| F01 | NI OVER 103% ________ SECS MAX________% |
| F02 | NI OVER 101.5% _______ SECS MAX________% |
| F03 | IGV MALFUNCTION |
| F04 | BLEED BAND NOT CLOSING PROPERLY |
| F06 | ENG OIL PRESS LOW |
| F07 | BRG 2 SCAV OIL TEMP HIGH |
| F08 | ENG OIL AND XMSN OIL TEMP HIGH |
| F09 | ENG OIL TEMP HIGH |
| F10 | ENG TORQ OVER 73 PSI ______ SECS MAX______PSI |
| F11 | ENG TORQ OVER 61 PSI ______ SECS MAX______PSI |
| F12 | ENG TORQ OVER 54 PSI ______ SECS MAX______PSI |
| F13 | ENG TORQ OVER 50 PSI ______ SECS MAX______PSI |
| F14 | N2 OVER 7180 ______SECS MAX______RPM |
| F15 | N2 OVER 6640 ______SECS MAX______RPM NI OVER 91% |
| F16 | N2 OVER 6800 ______SECS MAX______RPM |
| F17 | EGT OVER 760 DEG C______SECS DURING START MAX______DEG C |
| F18 | EGT OVER 675 DEG C______SECS DURING START MAX______DEG C |
| F19 | EGT OVER 760 DEG C______SECS MAX_________DEG C |
| F20 | XMSN OIL TEMP HIGH |
| F21 | XMSN OIL TEMP INST FAULT |
| F23 | XMSN OIL PRESS LOW |
| F24 | XMSN OIL PRESS INST FAULT |
| F25 | XMSN OIL PRESS HIGH |
| F26 | XMSN INT FILTER PRESS DROP HIGH |
| F27 | RPM WARN SYST NOT DETECTING UNDERSPEED |
| F28 | ROTOR OVER 356 RPM____SECS MAX______RPM |

TABLE 1 (Continued)

| Message No. | DA Printout |
|---|---|
| F29 | ROTOR OVER 339 RPM _____ SECS MAX______RPM |
| F30 | RPM WARN SYST NOT DETECTING OVERSPEED |
| F32 | EGT OVER 675 DEG C______SECS MAX______DEG C |
| F33 | EGT OVER 610 DEG C______MINUTES MAX_____DEG C |

AIDAPS Equipment or Sensor Failure

| Message No. | DA Printout | Meaning/Action |
|---|---|---|
| A00 | CMU FAILURE | ROM Fail |
| A01 | CMU FAILURE | ROM Fail |
| A03 | DAU FAILURE | DAU Self Test or No Inter |
| A04 | SERIAL NUMBERS DO NOT MATCH | |
| A11 | SENSOR FAILURE | VIB Sensor No. 1 Brg 45 |
| A12 | SENSOR FAILURE | VIB Sensor No. 2 Brg 2 |
| A13 | SENSOR FAILURE | VIB Sensor No. 3 Brg 1-21 |
| A14 | SENSOR FAILURE | VIB Sensor No. 4 Brg 3-4 |
| A15 | SENSOR FAILURE | VIB Sensor No. 5 Red Gr Ax |
| A16 | SENSOR FAILURE | VIB Sensor No. 6 XIQ |
| A17 | SENSOR FAILURE | VIB Sensor No. 7 XMB |
| A18 | SENSOR FAILURE | VIB Sensor No. 8 XRGR |
| A19 | SENSOR FAILURE | VIB Sensor No. 9 XTR |
| A20 | SENSOR FAILURE | VIB Sensor No. 10 HGB |
| A21 | SENSOR FAILURE | VIB Sensor No. 11 90 IQ |
| A22 | SENSOR FAILURE | VIB Sensor No. 12 42 IQ |
| A23 | SENSOR FAILURE | VIB Sensor No. 13 42 OQ |
| A31 | SENSOR FAILURE | EOT |
| A32 | SENSOR FAILURE | XOT |
| A34 | SENSOR FAILURE | B2DT |
| A35 | SENSOR FAILURE | IOFDP |
| A36 | SENSOR FAILURE | XOP |
| A38 | SENSOR FAILURE | N2 |
| A39 | SENSOR FAILURE | NR |

NOTE: Blanks in messages are automatically filled in by DA.

TABLE 1 (Continued)

| Message No. | DA Printout | Meaning/Action |
| --- | --- | --- |
| A40 | SENSOR FAILURE | SHP |
| A41 | SENSOR FAILURE | FT |
| A42 | SENSOR FAILURE | TI |
| A43 | SENSOR FAILURE | IGV |
| A44 | SENSOR FAILURE | FP |

## TABLE 2

## DIAGNOSTIC ANALYZER GAS PATH MESSAGES

| Message No. | Meaning |
|---|---|
| E01 | Significant engine performance change detected |
| E02 | Sensor malfunction |
| ENG FAULT 42 | Significant engine performance change (see other fault messages for diagnosis) |
| ENG FAULT 45 | EGT spread out of limits |
| ENG FAULT 60 | Excessive bleed during anti-icing indicated |
| ENG FAULT 66 | Compressor degradation |
| ENG FAULT 67 | Turbine efficiency loss may be due to fuel nozzle clogging/erosion |
| ENG FAULT 68 | Gas producer turbine degradation |
| ENG FAULT 69 | Power turbine degradation |
| ENG FAULT 70 | Gas producer turbine nozzle degradation |
| ENG FAULT 71 | Significant turbine inlet temperature increase due to indicated component degradation. |
| ENG FAULT 72 | Significant turbine inlet temperature increase (see numerical results) |
| ENG FAULT 73 | Significant airflow increase may be due to IGV schedule change |

TABLE 3

SUPPLEMENTARY SENSOR/SYSTEM ADVISORY MESSAGES

| Message No. | Reference* | Meaning |
|---|---|---|
| GPMSG 6 | C | Mode 3--if no other message (7 through 16) check engine stability |
| GPMSG 7 | C | Mode 3--check NI and TI |
| GPMSG 8 | C | Mode 3--check EGT |
| GPMSG 9 | C | Mode 3--check TORQUE and N2 |
| GPMSG 10 | D | Excessive number of Mode 2 data sets |
| GPMSG 11 | E | NI credibility (NIC > 101.5 pct) |
| GPMSG 12 | E | NI or TI credibility (NIC > military power limit +0.5 pct |
| GPMSG 14 | E | TI credibility (TI > $49^{9}$C or TI < $-23^{\circ}$C) |
| GPMSG 15 | E | N2 credibility (N2 > 101.5 pct or N2 < 96 pct) |
| GPMSG 16 | E | EGT calculated average vs measured average differ by more than $5^{\circ}$C |
| GPMSG 17 | E | T3 credibility (T3A and T3B difference > $20^{\circ}$C) |
| GPMSG 18 | E | PI credibility (PI < 10.25 psi) |
| GPMSG 19 | E | PI credibility (PI > 14.5 psi) |
| GPMSG 20 | E | Bleed band on signal |
| GPMSG 21 | E | Heater on with TI > $20^{\circ}$C (reset to off) |
| GPMSG 22 | E | Heater off with TI < $5^{\circ}$C |
| GPMSG 23 | E | Anti-icing on with TI > $5^{\circ}$C |
| GPMSG 35 | F | N2 or TORQUE out of range |
| GPMSG 36 | F | Fuel temperature incredible. Replaced by TI. |
| GPMSG 37 | G | NIC range for baseline insufficient (less than 4 pct) |

TABLE 3 (Continued)

| Message No. | Reference* | Meaning |
|---|---|---|
| GPMSG 38 | G | Nominal baselines used |
| GPMSG 39 | G | Interim baselines used |
| GPMSG 43 | H | Excessive gas path parameter variation from interim baseline |
| GPMSG 44 | H | Excessive fuel pressure variation from interim baseline |
| GPMSG 46 | H | Excessive EGT spread variation from interim baseline |
| GPMSG 47 | H | Insufficient data for engine performance checks (less than three samples) |
| GPMSG 48 | H | Flight logic results not verified |
| GPMSG 49 | I | P4C variation from BL incredible (> 10 pct) |
| GPMSG 50 | I | T3C variation from BL incredible (> 5 pct) |
| GPMSG 51 | I | WFC variation from BL incredible (> 20 pct) |
| GPMSG 52 | I | SHPC variation from BL incredible (> 25 pct) |
| GPMSG 53 | I | EGTC variation from BL incredible (> 10 pct) |
| GPMSG 54 | I | P1 signal malfunction indicated |
| GPMSG 55 | I | T1 signal malfunction indicated |
| GPMSG 56 | I | N1 signal malfunction indicated |
| GPMSG 59 | I | EGT spread variation incredible (> 30 pct) |
| GPMSG 61 | I | Fuel pressure variation incredible (> 50 pct) |
| GPMSG 98 | B | Low confidence level for results (ENG FAULT messages 66 through 73) |
| GPMSG 99 | G | Insufficient data for baseline update |

*See Reference 6, Appendix M "Operating Instructions for AIDAPS Gas Path Programs".

# APPENDIX C

# TROUBLE REPORT SUMMARY

TROUBLE REPORT (TR) SUMMARY - AIRCRAFT WIRING AND INSTALLATION RELATED

| Date | T.R. No. | Aircraft No. | Failed Item |
|---|---|---|---|
| 5-12-76 | 29042 | BC14 | IGV Coupling |
| Description<br>Broken during engine change. | | | |
| | | | |
| 5-19-76 | 29048 | BC-8 | IGV Coupling clamp |
| Clamp broken during engine change. | | | |
| | | | |
| 5-10-76 | 29037 | BC-12 | $T_3B$ Thermocouple Cable |
| GPMSG17 - Found severely crushed wire. | | | |
| Replaced. | | | |
| 5-12-76 | 29041 | BC-13 | $T_3A$ Thermocouple Cable |
| GPMSG17 - Found wire broken at transducer terminal. | | | |
| Repaired. | | | |
| 5-21-76 | 29046 | BC-13 | $T_3B$ Thermocouple Cable |
| GPMSG17 - Conductor separated about 8" from $T_3B$ Probe - evidence of bruising. | | | |
| Replaced cable. | | | |
| 8-2-76 | 29071 | BC-12 | $T_3A$ Thermocouple Cable |
| Preflight check - found open wire. | | | |
| Replaced. | | | |

| Date | T.R. No. | Aircraft No. | Failed Item |
|---|---|---|---|
| 7-23-76 | 29067 | BC-13 | $T_3A$ Thermocouple Cable |
| Description | | | |
| GPMSG17 - Wires to $T_3A$ broken. | | | |
| Repaired. | | | |
| 9-14-76 | 29082 | BC-14 | $T_3A$ and $T_3B$ Cables |
| GPMSG17 - Shorts in cables. | | | |
| Replaced both cables. | | | |
| 9-14-76 | 29083 | BC-12 | $T_3A$ and $T_3B$ Cables |
| During check $T_3A$ and $T_3B$ intermittently shorted - moving cables shorts appear and disappear. | | | |
| Replaced cables. | | | |
| 9-15-76 | 29064 | BC-13 | $T_3A$ and $T_3B$ Cables |
| Broken wires after flight near terminals. | | | |
| Repaired. | | | |
| 9-15-76 | 29062 | BC-13 | $T_3A$ Cable |
| Broken wire after flight. | | | |
| Repaired. | | | |
| 7-28-76 | 29289 | BC-13 | $T_3A$ Cable Lug |
| GPMSG17 and E02 - Probe lug found touching case. | | | |
| Repaired. | | | |

| Date | T.R. No. | Aircraft No. | Failed Item |
|---|---|---|---|
| 5-10-76 | 29040 | BC13 Sensor No. 2 Cable | Accel. Cable |
| Description<br>Pre-Flight - Wire to bearing No. 3/4 Sensor excessively frayed. | | | |
| Replaced cable. | | | |
| 7-12-76 | 29059 | BC13 | Cable to EGT Isolator |
| GPMSG16 - Found wires pulled out of connector pins at EGT isolator mating connector. | | | |
| Repaired. | | | |
| 7-26-76 | 29068 | BC8 - CDP | Engine Tubing |
| GPMSG49 - CDP Data variation - broken line to start fuel nozzle purge. | | | |
| | | | |
| 5-28-76 | 29052 | BC-14 Sensor No. 4 | Accel. Cable |
| Preflight check found frayed cable at connector. | | | |
| Replaced. | | | |
| 7-1-76 | 29054 | BC-14 Xmsn Int Oil $\Delta P$ | Connector Seal |
| Message A35 - Found water in connector. | | | |
| | | | |
| 9-9-76 | 29079 | BC-14 T3A | Connector Seal |
| Preflight - $T_3A$ data indicated short - found water in connector. | | | |
| | | | |

| Date | T.R. No. | Aircraft No. | Failed Item |
|---|---|---|---|
| 9-10-76 | 29080 | BC-14 HB No. 4 | Accel. Cable |
| Description<br>Found HB No. 4 Cable chafed. | | | |
| Replaced. | | | |
| 7-28-76 | 29072 | BC-13 Torque Pressure | Gulton 3135-7708 S/N 1004 |
| GPMSC 35 - Found pressure lines to transducer reversed. | | | |
| Calibration of sensor changed by reverse pressure. | | | |
| 2-10-76 | 29022 | BC-12 Torque Pressure | Gulton 3135-7708 S/N 1008 |
| Data showed torque too low. Found pressure lines to transducer reversed. | | | |
| Calibration of sensor changed. | | | |
| 5-19-76 | 29047 | BC-8 | Bleed Band Switch BZ-2RQ1244A-2 |
| Broken during engine harness change. | | | |
| | | | |
| 9-20-76 | 29089 | BC-13 Bleed Band Switch | BZ-2RQ1244A-2 S/N 7534 |
| Switch broken during engine change. | | | |
| | | | |
| 5-10-76 | 29039 | BC14 $T_1$ | 56BP17B S/N 0040 |
| GPMSG14 - Found keyway and pins sheared off. Mating connector overtorqued. | | | |
| | | | |

TROUBLE REPORT (TR) SUMMARY - AIRCRAFT WIRING AND INSTALLATION RELATED

| Date | T.R. No. | Aircraft No. | Failed Item |
|---|---|---|---|
| 9-14-76 | 29075 | BC-8 Sensor No. 7 | 6222M26 Accel. S/N BZ99 |
| Description<br>Sensor installed improperly - suffered mechanical damage. | | | |
| Removed from A/C. | | | |
| 11-1-76 | 29299 | BC-12 Sensor No. 13 | 6222M26 Accel. Mounting |
| Data analysis indicated sensor problem - frequency response change - found two loose mounting screws and third screw oversize. | | | |
| Remounted with proper screws. | | | |
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| Date | T.R. No. | Aircraft No. | Failed Item |
|---|---|---|---|
| 4-19-76 | 29034 | BC-12 Sensor No. 7 | 6222M26 Accel. S/N BW19 |
| Description<br>Data printout indicated sensor failure. Troubleshoot shows no output from sensor. Vendor conformed failure. Found broken wire at weld to crystal tab. Defective lead wire considered most probable. | | | |
| Unit removed from A/C. | | | |
| 5-5-76 | 29035 | BC-13 Sensor No. 11 | 6222M26 Accel. S/N CE82 |
| Data printout (DI) indicated data low amplitude. Message "A21". Vendor could not conform problem. | | | |
| Unit removed from A/C. Recalibrated and returned unit. | | | |
| 7-8-76 | 29057 | BC-12 Sensor No. 9 | 6222M26 Accel. S/N CC-68 |
| Message A19. Data indicated sensor failed. Unit has abrasive marks on dome. | | | |
| Unit removed from A/C. | | | |
| 7-8-76 | 29058 | BC-14 Sensor No. 7 | 6222M26 Accel. S/N C A01 |
| Message A17. Data indicated sensor failed. | | | |
| Unit removed from A/C and returned to vendor. Vendor unable to conform failure. | | | |
| 9-10-76 | 29085 | BC-8 $42^{o}$ G/B | 6222M26 Accel. S/N CC36 |
| Continuity check - Short Pin A to case. | | | |
| Unit removed from A/C and returned to vendor for analysis - short conformed. | | | |
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TROUBLE REPORT (TR) SUMMARY - SENSORS

| Date | T.R. No. | Aircraft No. | Failed Item |
|---|---|---|---|
| 9-14-76 | 29076 | BC-13 Sensor No. 13 | 6222M26 Accel. S/N BZ78 |
| Description<br>Data indicated failed sensor. Continuity check - Unit shorted Pin B to Case. | | | |
| Unit removed from A/C. Vendor conformed short. | | | |
| 9-14-76 | 29077 | BC-13 Sensor No. 5 | 6222M26 Accel. S/N CB96 |
| Data indicated sensor failing. Continuity check - Unit shorted Pin B to case. | | | |
| Unit removed from A/C. Vendor conformed short. | | | |
| 9-14-76 | 29078 | BC-12 Sensor No. 8 | 6222M26 Accel. S/N CC46 |
| Continuity check - Short Pin A to Case. | | | |
| Unit removed from A/C. Vendor conformed short. | | | |
| 9-15-76 | 29084 | BC-12 Sensor No. 13 | 6222M26 Accel. S/N CG45 |
| Data analysis at Torrance indicated failed sensor. | | | |
| Replaced unit. Vendor conformed failure - Change in frequency response. | | | |
| 10-21-76 | 29093 | BC-14 Sensor No. 8 | 6222M26 S/N CA74 |
| Data indicated bad sensor. | | | |
| Replaced. | | | |
| 10-26-76 | 29094 | BC-14 Hanger Bearing | 6222M26 S/N CC14 |
| Message A-20. | | | |
| Replaced. | | | |

## TROUBLE REPORT (TR) SUMMARY - SENSORS

| Date | T.R. No. | Aircraft No. | Failed Item |
|---|---|---|---|
| 9-15-76 | 29086 | BC-8 Sensor No. 4 | 6233 Accel. S/N BY13 |
| Description<br>Data Analysis at Torrance indicates unit may be causing false alarms. | | | |
| Replaced. Vendor could not conform failure. | | | |
| 9-28-76 | 29090 | BC-14 Sensor No. 2 | 6233 Accel. SN BX41 |
| Message A-12. | | | |
| Replaced. | | | |
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TROUBLE REPORT (TR) SUMMARY - SENSORS

| Date | T.R. No. | Aircraft No. | Failed Item |
|---|---|---|---|
| 1-20-76 | 29019 | BC-8 CDT No. 2 | HAC-15753 S/N R8669 |
| Description GPMSG17 - Intermittent Open Thermocouple. Vendor Analysis - Fractured Alumel wire at point where wire emerges from insulation; most probable cause is high vibration. | | | |
| | | | |
| 3-2-76 | 29026 | BC-12 | HAC-15753, S/N R8675 |
| $T_3$ Data erratic - Intermittent open Thermocouple. Vendor Analysis - Fractured Alumel wire at point where wire emerges from insulation; most probable cause is high vibration. | | | |
| | | | |
| 5-10-76 | 29038 | BC-14 $T_3A$ | HAC-15753, S/N R8676 |
| GPMSG17 - T3A Sensor Thermocouple Open. Vendor Analysis - Fractured Alumel Wire at point where wire emerges from insulation - Most probable cause is high vibration. | | | |
| | | | |
| 7-29-76 | 29070 | BC-13, T3A | HAC-15753, S/N R8670 |
| GPMSG17 - Replaced Sensor - Thermocouple Open - Vendor Analysis - fractured Alumel wire at point where wire emerges from insulation - Most probable cause is high vibration. | | | |
| | | | |
| 9-14-76 | 29087 | BC-12, T3B | HAC-15753, S/N R8673 |
| GPMSG17 - Data indicated T3B reading low. | | | |
| Replaced sensor. Could not conform failure at AiResearch. | | | |
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| Date | T.R. No. | Aircraft No. | Failed Item |
|---|---|---|---|
| 2-2-76 | 29020 | BC8 $T_1$ | 56BP17B S/N 00047 |
| Description GPMSG14 - Intermittent Open | | | |
| At room temp probe open. | | | |
| 3-2-76 | 29032 | BC-8 $T_1$ | 56BP17B S/N 00018 |
| Data drifting up. | | | |
| Resistance element grounded to case. | | | |
| 5-21-76 | 29049 | BC-12 $T_1$ | 56BP17B S/N 00043 |
| Preflight test, $T_1$ channel read full scale. | | | |
| Probe found open. | | | |
| 9-20-76 | 29088 | BC14 $T_1$ | 56BP17B S/N 00041 |
| GPMSG-14 Replaced Sensor. | | | |
| Temp probe found open. | | | |
| 8-17-76 | 29081 | BC14 $T_1$ | 56BP17B S/N 00051 |
| $T_1$ probe Shift - GPMSG 66 | | | |
| Lab test - Unit functional; may be slightly high. | | | |
| 8-23-76 | 29290 | BC 14 $T_1$ | 56BP17B S/N XXX |
| GPMSG14 - replaced probe. | | | |
| | | | |

| Date | T.R. No. | Aircraft No. | Failed Item |
|---|---|---|---|
| 3-25-76 | 29027 | BC-13 | BZ-2RQ1244A-2 Bleed Band Switch |
| Description Switch plunger loose. | | | |
| Repaired. | | | |
| 4-16-76 | 29031 | BC-12 Bearing No. 2 Scav. Oil Press. | Celesco PL 6-50-A2 S/N 2064 |
| Message A33 - Resistance check found open strain gage circuit. Vendor: Interconnect lead broken and resistor lead broken, apparently due to excessive vibration. | | | |
| | | | |
| 7-14-76 | 29063 | BC-13 XMSN Int Oil Filt ΔP | TRU-66A S/N 62868 |
| Message A-35, Transducer had 5 psi offset. | | | |
| Replaced. This is a GFE sensor. | | | |
| 2-10-76 | 29025 | BC-12 | 2101304-1-1 EGT Isolator |
| Erratic EGT Indicator Readings - found broken solder connection in EGT Isolator. | | | |
| Repaired. | | | |
| 7-7-76 | 29056 | BC-08 EGT Isolator | 2101304-1-1 S/N 85-D4 |
| GPMSG16 - EGT Channel 3 and 4, 5 and 6 Read high. Traced to isolator. | | | |
| Replaced. | | | |
| | | | |
| | | | |
| | | | |

| Date | T.R. No. | Aircraft No. | Failed Item |
|---|---|---|---|
| 8-9-76 | 29074 | - | 2101042, S/N 85-D2 |
| Description Recorder did not operate properly on ground. Recorded only three data blocks on flight--Worked properly on post flight ground run. | | | |
| Returned to vendor for repair - vendor found intermittent connection at transport connector (poor crimp). Repaired. | | | |
| 2-2-76 | 29021 | - | 2101042 S/N 85-D2 |
| Intermittent inflight and on bench - Confirmed problem at AiResearch-Torrance. | | | |
| Vendor found wire on wire wrap Pin shorting intermittently to case. Short stopped unit - Repaired. | | | |
| 5-13-76 | 29044 | BC-13 | 2101042 S/N 85-D2 |
| Recorder had only 17 mm. Data after two-hour flight - couldn't find anything wrong - flew again, no problem. | | | |
| | | | |
| 6-14-76 | 29205 | - | 2101042 S/N 85-D2 |
| Unable to record and replay data, while other recorder worked properly in same test setup. Vendor found tape oxide coming off and sticking to head. | | | |
| Replaced tape and remachined head. | | | |
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| Date | T.R. No. | Aircraft No. | Failed Item |
|---|---|---|---|
| 3-17-76 | 29030 | - | 2101040, S/N 85-D2 |
| Description<br>No filtered N2 during normal flight. Found two frequency channels not jumpered together. | | | |
| Add jumpers, correct drawings. | | | |
| 5-5-76 | 29036 | - | 2101040, S/N 85-D2 |
| GPMSG16 - From Data EGT No. 7 and 8 Read high. | | | |
| Repaired cold solder joint (not correct fix). | | | |
| 5-19-76 | 29045 | BC-13 | 2101040 S/N 85-D2 |
| GPMSG16 - Trouble shooting indicated heat sensitive op amp - | | | |
| Replaced. (Not correct fix) | | | |
| 7-1-76 | 29053 | - | 2101040 S/N 85-D2 |
| GPMSG16 - Replaced op amp again. | | | |
| (Not correct fix). | | | |
| 7-12-76 | 29060 | - | 2101040 S/N 85-D2 |
| GPMSG16 - Replaced MUX Chip - Still had problem. | | | |
| Add hard wire around circuit board etch (not total fix). | | | |
| 7-15-76 | 29061 | BC-13 | 2101040 S/N 85-D2 |
| GPMSG16 - Add second jumper around circuit board etch. | | | |
| Problem corrected - Apparently fault was in the circuit board etch, and was heat sensitive--intermittent: This made fault isolation extremely difficult. The above four TR's are incorrect fixes of the same problem. | | | |

## TROUBLE REPORT (TR) SUMMARY - DAU (Continued)

| Date | T.R. No. | Aircraft No. | Failed Item |
|---|---|---|---|
| 8-23-76 | 29236 | - | 2101040 S/N 85-D1 |
| **Description** Many vibration fault messages - no coherent vib output from DAU - No output (saturated) from charge converter. | | | |
| Bad vib MUX chips - replaced - microscopic inspection of chips showed damage evidence of high voltage short duration overstress. | | | |
| 8-23-76 | 29234 | - | 2101040 S/N 85-D2 |
| Many vibration faults - Problem occurred simultaneously with TR 29236 - double output from charge converter indicated one bad MUX. | | | |
| Replaced both MUX chips - same microscopic inspection as TR 29236. | | | |
| 8-30-76 | 29206 | BC-14 | 2101040 S/N 85-D2 |
| Many vibration faults - same symptoms as above two TR's. | | | |
| Replaced both MUX chips - Unit tested ok. | | | |
| 9-22-76 | 29288 | BC-8 | 2101040 S/N 85-D1 |
| Many vib faults - Same symptoms as above three TR's. | | | |
| Static discharge was considered to be most likely cause - a protective fix was developed and installed. | | | |
| 10-18-76 | 29291 | BC-8, BC-12 | 2101040 S/N 85-D1 |
| Analysis of vibration data verified Channel 9 had been failed between 7-7-76 and 8-20-76, Action in TR 29236. | | | |
| Repaired problem. | | | |
| | | | |

| Date | T.R. No. | Aircraft No. | Failed Item |
|---|---|---|---|
| 3-17-76 | 29029 | - | 909724, S/N 75-101 Core Memory<br>927390-101F S/N 361494 |
| Description<br>Data in memory lost when not writing - replace memory - returned removed unit to AiResearch-Torrance | | | |
| Failed parts associated with V-Boost Circuit - Repaired. | | | |
| 6-3-76 | 29051 | - | 909724, S/N 75-101<br>+28 v Power Supply |
| 28 v PS Output Transient to ≈ 40 volts - light on DA Got very bright. | | | |
| Replace 28 v PS and, precautionarily, +15 v reg. | | | |
| 6-16-76 | 29295 | - | 909724 S/N 75-101 |
| DA operation deteriorating - (1) Occasional power turn-on problem--won't start, (2) Core memory loses data, (3) Reading of recorder data would terminate part way through tape - Returned to AiResearch. Findings - (1) Two pins shorted together on mother board. (2) +5 volt Power Supply would occasionally not start, (3) +15 v power would occasionally oscillate, (4) Core Memory had failed transistors in V-Boost Circuit. | | | |
| (1) Replace +5 Power Supply (2) Repair shorted pins - (3) modified +15v power supply to prevent oscillation, (4) Core memory was repaired and its design modified to prevent problem. | | | |
| 7-19-76 | 29065 | - | DA Console |
| Cannot interrogate program with console - Return console to AiResearch for repair. | | | |
| Console repaired. | | | |
| 7-21-76 | 29066 | - | 909724 S/N 75-101 |
| Printer will not respond to DA commands - Replace Core Memory. | | | |
| Core Memory repaired by vendor - V-Boost Circuit fix installed. | | | |

| Date | T.R. No. | Aircraft No. | Failed Item |
|---|---|---|---|
| 7-31-76 | 29073 | - | Paper Tape Reader 800184 |
| Description<br>Paper Tape Reader Bulb would not light. | | | |
| Power Transistor, Q7, failed - Replaced. | | | |
| 9-30-76 | 29091 | - | 909724 S/N 75-102 |
| Printer runs slow - Traced to $U_2$ on $A_7$ Board. | | | |
| Replaced IC | | | |
| 9-30-76 | 29092 | - | 909724 S/N 75-102 |
| Clock on A2 Board doesn't function properly. Returned Board to AiResearch. | | | |
| Found failed U37 on CPU No. 2 Board - Replaced. | | | |
| | | | |
| | | | |
| | | | |
| | | | |
| | | | |
| | | | |
| | | | |
| | | | |
| | | | |

## APPENDIX D

## AIDAPS MTBF CALCULATIONS

## AIDAPS MTBF CALCULATIONS

1. Total flight test hours (est.) = 754

   (January 1 through November 16)

2. The following assumptions are made for additional on-time:

   (a) Ground power on data acquisition unit (DAU)

   10 hr/week X 44 weeks = 440

   (b) Digital data recorder (DDR)

   Ground power 1 hr/flt. hr

   (c) Computer memory unit (CMU)

   Ground power 1/2 hr/flt. hr

   (d) Diagnostic analyzer (DA)

   8 hr/day X 5 day/week X 44 weeks = 1760

   (e) Aircraft operating time 2X test time

3. DAU

   DAU's are on all flights

   (a)

| | |
|---|---|
| Flight time | 754 |
| Ground power time | 440 |
| Total time | 1194 |

   (b) Four failures

   MTBF - 1194/hr/4 failures = 299 hr/failure

   (c) Three design deficiency failures

   (MUX Protection Fix, TR29288)

   MTBF(P) = 1194 hr/1 failure = 1194 hr/failure

4. DDR

Two-thirds of flight time with recorder

| | | |
|---|---|---|
| (a) | Flight time 2/3 X 754 | 503 |
| | Ground time 1/2 X 754 | 377 |
| | Total time | 880 |
| (b) | Three failures | |
| | MTBF = 880 hr/3 failures = | 293 hr/failure |

5. CMU

One-third of flight time with CMU

| | | |
|---|---|---|
| (a) | Flight time 1/3 X 754 | 251 |
| | Ground time 1/2 X 754 X 1/2 | 125 |
| | Total time | 376 |
| (b) | No failures | |
| | MTBF | > 376 |

6. Sensors

Two-hr power-on per flight-hr X flight time X No. of added sensors

| | | |
|---|---|---|
| (a) | 2 x 754 X 29 | 43,732 |
| (b) | 29 failures | |
| | MTBF - 43,732 hr/29 failures = | 1,508 hr/failure |
| (c) | 11 non-design deficiency failures | |
| | MTBF = 43,732 hr/11 failures = | 3,975 hr/failure |

7. Aircraft wiring and installation related

Two-hr power-on per flight hr X flight time X No. of monitored signals

| | | |
|---|---|---|
| (a) | Operating Hours | |
| | 2 X 754 X 54 | 81,432 |
| (b) | 26 failures | |
| | MTBF = 81,432 hr/26 failures = | 3,132 |

(c) 13 non-design-related failures

MTBF(P) = 81,432 hr/13 = 6,264

8. DA

Assume operation 40 hr per week

(a) Operating hours

40 hr X 44 weeks = 1,760

(b) Six failures

MTBF = 1,760/6 = 293

(c) 2 non-design deficiencies

MTBF = 1,760/2 = 880

# APPENDIX E

## AIDAPS VIBRATION LOGIC DETAILS SUMMARY

# AIDAPS VIBRATION LOGIC DETAILS

SENSOR NO: 1  LOCATION: Engine Gearbox

TYPE: Endevco 6222M26

COMPONENT PARTS MONITORED: No. 45 Bearing

Bearing Fault Detection Bands

| Band No. | Center Freq, Hz | Band-Width |
|---|---|---|
| 1 | 2966 | 312 |
| 2 | 3284 | 312 |
| 3 | 3589 | 312 |
| 4 | 13281 | 312 |
| 5 | 13599 | 312 |
| 6 | 13904 | 312 |
| 7 | 14221 | 312 |
| 8 | 14526 | 312 |
| 9 | 189[illegible] | 312 |
| 10 | 19214 | 312 |
| 11 | 19531 | 312 |
| 12 | 23596 | 312 |
| 13 | 23901 | 312 |
| 14 | 24219 | 312 |
| 15 | 24536 | 312 |
| 16 | 24841 | 312 |
| 17 | 27966 | 312 |
| 18 | 28284 | 312 |
| 19 | 28589 | 312 |
| 20 | 28906 | 312 |
| 21 | 29224 | 312 |
| 22 | 29529 | 312 |
| 23 | 33911 | 312 |
| 24 | 34216 | 312 |
| 25 | 34534 | 312 |
| 26 | 34839 | 312 |
| 27 | 38904 | 312 |
| 28 | 39221 | 312 |
| 29 | 39526 | 312 |
| 30 | 39844 | 312 |

**Discrimination Index Limits**

Diagnostic: 3.0

Flight Safety: 7.0

## AIDAPS VIBRATION LOGIC DETAILS

SENSOR NO: 2 LOCATION: Engine Combustor

TYPE: Endevco 6233

COMPONENT PARTS MONITORED: No. 2, 3, 4 Bearings and Rotor Unbalance

Bearing Fault Detection Bands

| Band No. | Center Freq, Hz | Band-Width |
|---|---|---|
| 1 | 281 | 52 |
| 2 | 342 | 52 |
| 3 | 391 | 52 |
| 4 | 439 | 52 |
| 5 | 2319 | 52 |
| 6 | 2368 | 52 |
| 7 | 2417 | 52 |
| 8 | 2466 | 52 |
| 9 | 3870 | 52 |
| 10 | 3931 | 52 |
| 11 | 3979 | 52 |
| 12 | 4028 | 52 |
| 13 | 4077 | 52 |
| 14 | 5334 | 52 |
| 15 | 5383 | 52 |
| 16 | 5432 | 52 |
| 17 | 5481 | 52 |
| 18 | 5542 | 52 |
| 19 | 5896 | 52 |
| 20 | 5957 | 52 |
| 21 | 6006 | 52 |
| 22 | 6055 | 52 |
| 23 | 6116 | 52 |
| 24 | 6421 | 52 |
| 25 | 6470 | 52 |
| 26 | 6531 | 52 |
| 27 | 6580 | 52 |
| 28 | 6628 | 52 |

Rotor Unbalance Bands

| Band No. | Rotor | Center Freq*, Hz |
|---|---|---|
| Fundamental Rotor Freq. | | |
| 31 | N1 | 419 |
| 29 | N2 | 351 |
| First Harmonic Rotor Freq. | | |
| 32 | N1 | 838 |
| 30 | N2 | 702 |

*All rotor bandwidths are 52 Hz
Center freq. is for NX = 100%

Discrimination Index Limits

Diagnostic: 1.6

Flight Safety: 6.0

Discrimination Index Limits

Diagnostic: 2.5

Flight Safety: 6.0

## AIDAPS VIBRATION LOGIC DETAILS

SENSOR NO: 3 LOCATION: Engine Compressor

TYPE: Endevco 6233

COMPONENT PARTS MONITORED: No. 1 Bearing and Rotor Unbalance

**Bearing Fault Detection Bands**

| Band No. | Center Freq, Hz | Band-Width |
|---|---|---|
| 1 | 183 | 52 |
| 2 | 232 | 52 |
| 3 | 281 | 52 |
| 4 | 342 | 52 |
| 5 | 391 | 52 |
| 6 | 2466 | 52 |
| 7 | 2527 | 52 |
| 8 | 2576 | 52 |
| 9 | 2625 | 52 |
| 10 | 4810 | 52 |
| 11 | 4858 | 52 |
| 12 | 4919 | 52 |
| 13 | 4968 | 52 |
| 14 | 5017 | 52 |
| 15 | 5334 | 52 |
| 16 | 5383 | 52 |
| 17 | 5432 | 52 |
| 18 | 5481 | 52 |
| 19 | 5542 | 52 |
| 20 | 6006 | 52 |
| 21 | 6055 | 52 |
| 22 | 6116 | 52 |
| 23 | 6165 | 52 |
| 24 | 6213 | 52 |
| 25 | 6262 | 52 |

**Rotor Unbalance Bands**

| Band No. | Rotor | Center Freq*, Hz |
|---|---|---|
| Fundamental Rotor Freq. | | |
| 28 | N1 | 419 |
| 26 | N2 | 351 |
| First Harmonic Rotor Freq. | | |
| 29 | N1 | 838 |
| 27 | N2 | 702 |

*All rotor bandwidths are 52 Hz
Center freq. is for NX = 100%

**Discrimination Index Limits** (Bearing)

Diagnostic: 3.5

Flight Safety: 7.0

**Discrimination Index Limits** (Rotor)

Diagnostic: 2.5

Flight Safety: 6.0

## AIDAPS VIBRATION LOGIC DETAILS

SENSOR NO: 4 LOCATION: Engine Power Turbine

TYPE: Endevco 6233

COMPONENT PARTS MONITORED: No. 3, 4 Bearing

Bearing Fault Detection Bands

| Band No. | Center Freq, Hz | Band-Width |
|---|---|---|
| 1 | 232 | 52 |
| 2 | 281 | 52 |
| 3 | 342 | 52 |
| 4 | 391 | 52 |
| 5 | 439 | 52 |
| 6 | 2161 | 52 |
| 7 | 2209 | 52 |
| 8 | 2258 | 52 |
| 9 | 2319 | 52 |
| 10 | 2368 | 52 |
| 11 | 2942 | 52 |
| 12 | 2991 | 52 |
| 13 | 3040 | 52 |
| 14 | 3088 | 52 |
| 15 | 3149 | 52 |
| 16 | 3198 | 52 |
| 17 | 3977 | 52 |
| 18 | 4028 | 52 |
| 19 | 4077 | 52 |
| 20 | 4138 | 52 |
| 21 | 4187 | 52 |
| 22 | 4761 | 52 |
| 23 | 4810 | 52 |
| 24 | 4858 | 52 |
| 25 | 4919 | 52 |
| 26 | 4968 | 52 |
| 27 | 6470 | 52 |
| 28 | 6531 | 52 |
| 29 | 6580 | 52 |
| 30 | 6628 | 52 |

Discrimination Index Limits

Diagnostic: 2.5

Flight Safety: 6.0

## AIDAPS VIBRATION LOGIC DETAILS

SENSOR NO: 5 LOCATION: Engine Gear Box, Axial

TYPE: Endevco 6222M26

COMPONENT PARTS MONITORED: No. 2 Bearing

Bearing Fault Detection Bands

| Band No. | Center Freq, Hz | Band-Width |
|---|---|---|
| 1 | 2661 | 312 |
| 2 | 2966 | 312 |
| 3 | 3284 | 312 |
| 4 | 3589 | 312 |
| 5 | 3906 | 312 |
| 6 | 11719 | 312 |
| 7 | 12036 | 312 |
| 8 | 12341 | 312 |
| 9 | 12659 | 312 |
| 10 | 12964 | 312 |
| 11 | 13281 | 312 |
| 12 | 13599 | 312 |
| 13 | 18591 | 312 |
| 14 | 18909 | 312 |
| 15 | 19214 | 312 |
| 16 | 19531 | 312 |
| 17 | 19849 | 312 |
| 18 | 20154 | 312 |
| 19 | 25781 | 312 |
| 20 | 26099 | 312 |
| 21 | 26404 | 312 |
| 22 | 26721 | 312 |
| 23 | 27026 | 312 |
| 24 | 34216 | 312 |
| 25 | 34534 | 312 |
| 26 | 34839 | 312 |
| 27 | 35156 | 312 |
| 28 | 35474 | 312 |
| 29 | 35779 | 312 |

Discrimination Index Limits

Diagnostic: 3.0

Flight Safety: 7.0

## AIDAPS VIBRATION LOGIC DETAILS

SENSOR NO: 6 LOCATION: Transmission Input Quill

TYPE: Endevco 6222M26

COMPONENT PARTS MONITORED: Input Quill and Planetary Bearings and Pinion and Ring Gears

Bearing Fault Detection Bands

| Band No. | Center Freq, Hz | Band-Width |
|---|---|---|
| 1 | 1721 | 312 |
| 2 | 2966 | 312 |
| 3 | 3284 | 312 |
| 4 | 3589 | 312 |
| 5 | 3906 | 312 |
| 6 | 7349 | 312 |
| 7 | 7654 | 312 |
| 8 | 7971 | 312 |
| 9 | 11401 | 312 |
| 10 | 18591 | 312 |
| 11 | 18909 | 312 |
| 12 | 19214 | 312 |
| 13 | 19531 | 312 |
| 14 | 22656 | 312 |
| 15 | 22974 | 312 |
| 16 | 23279 | 312 |
| 17 | 23596 | 312 |
| 18 | 24536 | 312 |
| 19 | 24841 | 312 |
| 20 | 25159 | 312 |
| 21 | 25464 | 312 |
| 22 | 26099 | 312 |
| 23 | 26404 | 312 |
| 24 | 26721 | 312 |
| 25 | 35779 | 312 |
| 26 | 36096 | 312 |
| 27 | 36401 | 312 |
| 28 | 36719 | 312 |
| 29 | 38904 | 312 |
| 30 | 39221 | 312 |
| 31 | 39526 | 312 |
| 32 | 39844 | 312 |

Discrimination Index Limits

Diagnostic: 2.0

Flight Safety: 6.0

Gear Fault Detection Bands

| Band No. | Side-Band No. | Center Freq*, Hz |
|---|---|---|
| Fundamental Gear Mesh | | |
| 33 | -5 | 5797 |
| 34 | -4 | 5907 |
| 35 | -3 | 6016 |
| 36 | -2 | 6126 |
| 37 | -1 | 6235 |
| 43 | 0 | 6344 |
| 38 | 1 | 6454 |
| 39 | 2 | 6563 |
| 40 | 3 | 6673 |
| 41 | 4 | 6782 |
| 42 | 5 | 6891 |
| First Harmonic Gear Mesh | | |
| 44 | -5 | 8970 |
| 45 | -4 | 9079 |
| 46 | -3 | 9188 |
| 47 | -2 | 9298 |
| 48 | -1 | 9407 |
| 54 | 0 | 9517 |
| 49 | 1 | 9626 |
| 50 | 2 | 9735 |
| 51 | 3 | 9845 |
| 52 | 4 | 9954 |
| 53 | 5 | 10063 |

*All gear bandwidths are 20 Hz
Center freq. is for NZ = 100%

Discrimination Index Limits

Diagnostic: 2.0

Flight Safety: 6.0

# AIDAPS VIBRATION LOGIC DETAILS

SENSOR NO: 7 LOCATION: Transmission Main Mast

TYPE: Endevco 6222M26

COMPONENT PARTS MONITORED: Mast Bearing

Bearing Fault Detection Bands

| Band No. | Center Freq, Hz | Band-Width |
|---|---|---|
| 1 | 3284 | 312 |
| 2 | 4846 | 312 |
| 3 | 5151 | 312 |
| 4 | 5469 | 312 |
| 5 | 5786 | 312 |
| 6 | 6091 | 312 |
| 7 | 10474 | 312 |
| 8 | 10779 | 312 |
| 9 | 11096 | 312 |
| 10 | 11401 | 312 |
| 11 | 17969 | 312 |
| 12 | 18286 | 312 |
| 13 | 18591 | 312 |
| 14 | 18909 | 312 |
| 15 | 19214 | 312 |
| 16 | 23279 | 312 |
| 17 | 23596 | 312 |
| 18 | 23901 | 312 |
| 19 | 24219 | 312 |
| 20 | 24536 | 312 |
| 21 | 27026 | 312 |
| 22 | 27344 | 312 |
| 23 | 27661 | 312 |
| 24 | 27966 | 312 |
| 25 | 33594 | 312 |
| 26 | 33911 | 312 |
| 27 | 34216 | 312 |
| 28 | 34534 | 312 |
| 29 | 35779 | 312 |
| 30 | 36096 | 312 |
| 31 | 36401 | 312 |
| 32 | 36719 | 312 |

Discrimination Index Limits

Diagnostic: 3.7

Flight Safety: 8.0

## AIDAPS VIBRATION LOGIC DETAILS

SENSOR NO: 8 LOCATION: Transmission Planetary Housing

TYPE: Endevco 6222M26

COMPONENT PARTS MONITORED: Planetary Sun Gears

Gear Fault Detection Bands

| Band No. | Side-Band No. | Center Freq*, Hz |
|---|---|---|
| Fundamental Gear Mesh | | |
| 1 | -5 | 945 |
| 2 | -4 | 1011 |
| 3 | -3 | 1077 |
| 4 | -2 | 1143 |
| 5 | -1 | 1209 |
| 11 | 0 | 1275 |
| 6 | 1 | 1341 |
| 7 | 2 | 1407 |
| 8 | 3 | 1473 |
| 9 | 4 | 1539 |
| 10 | 5 | 1605 |
| First Harmonic Gear Mesh | | |
| 12 | -5 | 1583 |
| 13 | -4 | 1649 |
| 14 | -3 | 1715 |
| 15 | -2 | 1781 |
| 16 | -1 | 1847 |
| 22 | 0 | 1913 |
| 17 | 1 | 1979 |
| 18 | 2 | 2045 |
| 19 | 3 | 2111 |
| 20 | 4 | 2177 |
| 21 | 5 | 2242 |

*All gear bandwidths are 20 Hz
Center freq. is for NZ = 100%

Discrimination Index Limits

Diagnostic: 3.0

Flight Safety: 7.0

## AIDAPS VIBRATION LOGIC DETAILS

SENSOR NO: 9 LOCATION: Transmission Tail Rotor Quill

TYPE: Endevco 6222M26

COMPONENT PARTS MONITORED: Tail Rotor Quill Bearings

**Bearing Fault Detection Bands**

| Band No. | Center Freq, Hz | Bandwidth |
|---|---|---|
| 1 | 1721 | 312 |
| 2 | 2026 | 312 |
| 3 | 2344 | 312 |
| 4 | 3906 | 312 |
| 5 | 4224 | 312 |
| 6 | 4529 | 312 |
| 7 | 4846 | 312 |
| 8 | 11096 | 312 |
| 9 | 11401 | 312 |
| 10 | 11719 | 312 |
| 11 | 12036 | 312 |
| 12 | 12341 | 312 |
| 13 | 22339 | 312 |
| 14 | 22656 | 312 |
| 15 | 22974 | 312 |
| 16 | 23279 | 312 |
| 17 | 23596 | 312 |
| 18 | 23901 | 312 |
| 19 | 24219 | 312 |
| 20 | 25464 | 312 |
| 21 | 25781 | 312 |
| 22 | 26099 | 312 |
| 23 | 29529 | 312 |
| 24 | 29846 | 312 |
| 25 | 30151 | 312 |
| 26 | 30469 | 312 |
| 27 | 30786 | 312 |
| 28 | 37659 | 312 |
| 29 | 37964 | 312 |
| 30 | 38281 | 312 |
| 31 | 38599 | 312 |
| 32 | 38904 | 312 |

**Discrimination Index Limits**

Diagnostic: 2.5

Flight Safety: 6.0

## AIDAPS VIBRATION LOGIC DETAILS

SENSOR NO: 10 LOCATION: Hanger Bearing No. 4

TYPE: Endevco 6222M26

COMPONENT PARTS MONITORED: Hanger Bearing

Bearing Fault Detection Bands

| Band No. | Center Freq, Hz | Bandwidth |
|---|---|---|
| 1 | 14001 | 4000 |
| 2 | 18005 | 4000 |
| 3 | 21997 | 4000 |
| 4 | 26001 | 4000 |
| 5 | 30005 | 4000 |
| 6 | 33997 | 4000 |
| 7 | 38000 | 4000 |

Discrimination Index Limits

Diagnostic: 7.0

Flight Safety: 25.0

# AIDAPS VIBRATION LOGIC DETAILS

SENSOR NO: 11 LOCATION: $90^{\circ}$ Gear Box

TYPE: Endevco 6222M26

COMPONENT PARTS MONITORED: All Bearings and Gears

Bearing Fault Detection Bands

| Band No. | Center Freq, Hz | Band-Width |
|---|---|---|
| 1 | 781 | 312 |
| 2 | 1099 | 312 |
| 3 | 3284 | 312 |
| 4 | 3589 | 312 |
| 5 | 3906 | 312 |
| 6 | 11401 | 312 |
| 7 | 11719 | 312 |
| 8 | 12036 | 312 |
| 9 | 12341 | 312 |
| 10 | 12659 | 312 |
| 11 | 18909 | 312 |
| 12 | 19214 | 312 |
| 13 | 19531 | 312 |
| 14 | 28906 | 312 |
| 15 | 29224 | 312 |
| 16 | 29529 | 312 |
| 17 | 29846 | 312 |
| 18 | 30151 | 312 |
| 19 | 30469 | 312 |
| 20 | 31714 | 312 |
| 21 | 32031 | 312 |
| 22 | 32349 | 312 |
| 23 | 32654 | 312 |

Gear Fault Detection Bands

| Band No. | Side-Band No. | Center Freq*, Hz |
|---|---|---|
| Fundamental Gear Mesh | | |
| 24 | -5 | 1783 |
| 25 | -4 | 1854 |
| 26 | -3 | 1926 |
| 27 | -2 | 1997 |
| 28 | -1 | 2068 |
| 34 | 0 | 2140 |
| 29 | 1 | 2211 |
| 30 | 2 | 2282 |
| 31 | 3 | 2354 |
| 32 | 4 | 2425 |
| 33 | 5 | 2496 |
| First Harmonic Gear Mesh | | |
| 35 | -5 | 2853 |
| 36 | -4 | 2924 |
| 37 | -3 | 2995 |
| 38 | -2 | 3067 |
| 39 | -1 | 3138 |
| 45 | 0 | 3209 |
| 40 | 1 | 3281 |
| 41 | 2 | 3352 |
| 42 | 3 | 3423 |
| 43 | 4 | 3495 |
| 44 | 5 | 3566 |

*All gear bandwidths are 20 Hz
Center frequency is for NZ = 100%

Discrimination Index Limits

Diagnostic: 2.6

Flight Safety: 6.0

Discrimination Index Limits

Diagnostic: 3.5

Flight Safety: 7.0

# AIDAPS VIBRATION LOGIC DETAILS

SENSOR NO: 12 LOCATION: 42° Gear Box Input Quill

TYPE: Endevco 6222M26

COMPONENT PARTS MONITORED: All Bearings and Gears

**Bearing Fault Detection Bands**

| Band No. | Center Freq, Hz | Band-Width |
|---|---|---|
| 1 | 1099 | 312 |
| 2 | 1404 | 312 |
| 3 | 2026 | 312 |
| 4 | 5151 | 312 |
| 5 | 5469 | 312 |
| 6 | 5786 | 312 |
| 7 | 6091 | 312 |
| 8 | 10474 | 312 |
| 9 | 10779 | 312 |
| 10 | 11096 | 312 |
| 11 | 11401 | 312 |
| 12 | 11719 | 312 |
| 13 | 17029 | 312 |
| 14 | 17346 | 312 |
| 15 | 17651 | 312 |
| 16 | 19531 | 312 |
| 17 | 19849 | 312 |
| 18 | 20154 | 312 |
| 19 | 20471 | 312 |
| 20 | 22339 | 312 |
| 21 | 22656 | 312 |
| 22 | 22974 | 312 |
| 23 | 23279 | 312 |
| 24 | 23596 | 312 |
| 25 | 23901 | 312 |
| 26 | 24219 | 312 |
| 27 | 24536 | 312 |
| 28 | 28284 | 312 |
| 29 | 28589 | 312 |
| 30 | 28906 | 312 |
| 31 | 29224 | 312 |
| 32 | 29529 | 312 |

**Gear Fault Detection Bands**

| Band No. | Side-Band No. | Center Freq*, Hz |
|---|---|---|
| Fundamental Gear Mesh | | |
| 33 | -5 | 1569 |
| 34 | -4 | 1640 |
| 35 | -3 | 1711 |
| 36 | -2 | 1783 |
| 37 | -1 | 1854 |
| 43 | 0 | 1925 |
| 38 | 1 | 1997 |
| 39 | 2 | 2068 |
| 40 | 3 | 2139 |
| 41 | 4 | 2210 |
| 42 | 5 | 2282 |
| First Harmonic Gear Mesh | | |
| 44 | -5 | 3494 |
| 45 | -4 | 3565 |
| 46 | -3 | 3636 |
| 47 | -2 | 3708 |
| 48 | -1 | 3779 |
| 54 | 0 | 3850 |
| 49 | 1 | 3922 |
| 50 | 2 | 3993 |
| 51 | 3 | 4064 |
| 52 | 4 | 4136 |
| 53 | 5 | 4207 |

*All gear bandwidths are 20 Hz
Center freq. is for NZ = 100%

**Discrimination Index Limits** (Bearing)

Diagnostic: 3.5

Flight Safety: 7.0

**Discrimination Index Limits** (Gear)

Diagnostic: 1.8

Flight Safety: 4.0

# AIDAPS VIBRATION LOGIC DETAILS

SENSOR NO: 13 LOCATION: 42° Gear Box Output Quill

TYPE: Endevco 6222M26

COMPONENT PARTS MONITORED: All Bearings

Bearing Fault Detection Bands

| Band No. | Center Freq, Hz | Band-Width |
|---|---|---|
| 1 | 1099 | 312 |
| 2 | 1404 | 312 |
| 3 | 2026 | 312 |
| 4 | 5151 | 312 |
| 5 | 5469 | 312 |
| 6 | 5786 | 312 |
| 7 | 6091 | 312 |
| 8 | 10474 | 312 |
| 9 | 10779 | 312 |
| 10 | 11096 | 312 |
| 11 | 11401 | 312 |
| 12 | 11719 | 312 |
| 13 | 17029 | 312 |
| 14 | 17346 | 312 |
| 15 | 17651 | 312 |
| 16 | 19531 | 312 |
| 17 | 19849 | 312 |
| 18 | 20154 | 312 |
| 19 | 20471 | 312 |
| 20 | 22339 | 312 |
| 21 | 22656 | 312 |
| 22 | 22974 | 312 |
| 23 | 23279 | 312 |
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